SECURITY INFORMATION

c.2

UNCLASSIFIED CONFIDENTIAL Copy 6
RM E52G22

NACA RM E52G22

OCT 5 1953

NACA

RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE CHARACTERISTICS OF THE J47-25 TURBOJET ENGINE - DATA PRESENTATION

By Paul E. Renas and Emmert T. Jansen

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

CLASSIFICATION CHANGED

To UNCLASSIFIED

By authority of Nasa TPA 11 Date Effective 12-1-59

NB 2-1-60

CLASSIFIED DOCUMENT

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
September 29, 1953

CONFIDENTIAL

UNCLASSIFIED

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE CHARACTERISTICS OF THE J47-25 TURBOJET ENGINE - DATA PRESENTATION

By Paul E. Renas and Emmert T. Jansen

SUMMARY

An investigation was conducted in an altitude test chamber at the NACA Lewis laboratory to determine the altitude performance of the J47-25 turbojet engine operating with a fixed-area exhaust nozzle. Data were obtained over a range of engine-inlet Reynolds numbers corresponding to altitudes from 18,000 to 54,000 feet and flight Mach numbers from 0.50 to 1.10.

Reducing the engine-inlet Reynolds number resulted in a reduction in corrected air flow but had essentially no effect on corrected exhaust-gas total temperature, corrected fuel flow, and engine pumping characteristics for a range of Reynolds number indices from 0.80 to 0.30. The corrected jet thrust parameter generalized throughout the range of engine-inlet Reynolds numbers investigated.

At a given corrected engine speed with critical pressure ratio existing in the exhaust nozzle, increasing the engine-inlet ram-pressure ratio from 1.0 to 1.25 decreased the corrected exhaust-gas temperature. Further increases in ram-pressure ratio had no effect on the exhaust-gas temperature.

INTRODUCTION

An investigation was conducted in an NACA Lewis altitude chamber to determine the altitude performance characteristics of a J47-25 axial-flow turbojet engine over a range of engine-inlet Reynolds number indices corresponding to altitudes from 18,000 to 54,000 feet and flight Mach numbers from 0.50 to 1.10. In order to simplify the procedure in obtaining performance data and to make the data applicable to any flight condition, Reynolds number index $\frac{\delta_1}{\varphi_1\sqrt{\theta_1}}$, which is proportional to Reynolds number at a given corrected engine speed and is a function only

of engine-inlet total pressure and temperature, was used instead of various set altitudes and flight Mach number combinations (reference 1). By the technique just mentioned, the data obtained in this investigation may be used to obtain the performance of the engine at any flight condition for which critical flow exists in the exhaust nozzle. An example 2625 is included in the appendix to illustrate the method of obtaining conventional performance parameters for a given flight condition from the data such as presented herein.

In addition to the basic engine performance, data were obtained in which the effects of variation of engine-inlet conditions on exhaust-gas temperature and thrust were observed. These effects are of importance from the standpoint of aircraft take-off and day-to-day weather variations.

All performance data obtained in this investigation are presented in both graphical and tabular form.

APPARATUS

Engine

The J47-25 axial-flow turbojet engine used in this investigation has a twelve-stage compressor, eight tubular combustion chambers, and a single-stage turbine. The engine has a static sea-level thrust rating of 6060 pounds at the rated engine speed of 7950 rpm and an engine manufacturer's turbine-outlet temperature of 1245° F. The compressor air flow is approximately 104 pounds per second and compressor pressure ratio is 5.3 at rated sea-level conditions. A conical exhaust nozzle having an area of 298.5 square inches was installed on the engine. Operation of the engine with this nozzle produced an average tail-pipe total gas temperature of 1710° R (1250° F), which is based on NACA instrumentation at static sea-level conditions and rated engine speed of 7950 rpm. The maximum dimensions of the engine are a 37-inch diameter and a 144-inch over-all length excluding the cylindrical tail pipe and the exhaust nozzle. The total weight of the engine is 2653 pounds.

Installation

The altitude test chamber in which the engine was installed is 10 feet in diameter and 60 feet in length. The test chamber is divided into three sections separated by bulkheads: the air-inlet section, the engine compartment, and the exhaust section. The engine was mounted on a thrust-measuring bed. A front bulkhead, which incorporated a labyrinth seal around the forward end of the engine, provided for freedom of movement of the engine in an axial direction. A rear bulkhead was installed to act as a radiation shield and to prevent recirculation of the hot exhaust gases about the engine.

2625

Instrumentation

The location of the instrumentation stations before and after each of the principal components of the engine is shown in figure 1. Sketches showing the arrangement of the separate temperature and pressure probes within a given station are presented in figure 2. The total-pressure tubes at stations 1 and 9 were located at the centers of 24 and 6 equal areas, respectively. The thermocouples at stations 1, 3, 5, and 9 and the total-pressure tubes at stations 3 and 5 were located on approximately equal spacings. The instrumentation at the engine inlet (station 1) was used in calculating the altitude and flight Mach number correction factors θ, δ, and φ. (All symbols are defined in the appendix.) The pressure and temperature measurements at station 9 were used to calculate ideal or rake jet thrust and nozzle gas flow. Measured jet thrust was also determined from scale readings for each condition investigated. The atmospheric pressure surrounding the jet nozzle was measured by four lip static probes located in the exhaust portion of the chamber (station 0).

Fuel flow was measured by two rotameters connected in series and calibration of the rotameters was made with the type fuel used in this investigation (MIL-F-5624A, grade JP-4).

PROCEDURE

The inlet conditions were varied to correspond to Reynolds number indices from 0.15 to 0.80. For each inlet condition, the exhaust pressure was reduced to the minimum of the exhaust system with the engine operating at rated speed. The inlet temperature and pressure and the exhaust pressure were then maintained constant while data were taken over a range of engine speeds from rated speed to approximately the speed where the exhaust nozzle became unchoked. A summary of the operating conditions covered in this investigation is given in the following table:

Reynolds number index	Inlet total temperature ($^{\circ}$R)	Inlet total pressure (lb/sq ft)	Ram-pressure ratio
0.15	410	232	1.19
.2	410	315	1.48
.25	410	387	1.64
.3	410	465	1.34
.3	410	465	1.70
.4	467	739	1.35
.425	437	718	1.41
.5	467	923	1.95
.6	467	1108	2.14
.8	530	1740	1.70

The methods of calculation are given in the appendix.

PRESENTATION OF DATA

The simulated altitude performance data obtained in this investigation were corrected to NACA standard altitude conditions and are presented in table I. Generalization of data for various engine-inlet conditions corresponding to a given Reynolds number index requires that critical flow be established in the exhaust nozzle. The range of corrected engine speeds over which the exhaust nozzle of the engine was choked is shown in figure 3 for a range of Reynolds number indices corresponding to various altitudes and flight Mach numbers. At all altitudes, this minimum corrected engine speed at which choking occurred decreased approximately linearly from about 7600 rpm at a flight Mach number of 0.2 to about 5750 rpm at a flight Mach number of 1.10. The data of this report may be used to determine performance only at flight conditions in the choked region above this curve. 2625

In order to aid in determining the Reynolds number index corresponding to a given flight condition and thereby determine the engine performance at NACA standard altitude conditions from the generalized data presented, the values of δ, θ, φ, and Reynolds number index are given in table II for a wide range of flight conditions; 100 percent ram-pressure recovery was assumed.

Effect of Engine-Inlet Conditions on Performance

In addition to the basic engine performance, two effects of special concern regarding exhaust-nozzle sizing and aircraft take-off are the effect of engine-inlet temperature on exhaust-gas temperature at sea-level static-pressure conditions and the effect of engine-inlet ram-pressure ratio on exhaust-gas temperature and thrust at low flight speeds and low altitudes. However, because of test-facility limitations, these effects had to be investigated at altitudes of 15,000 and 20,000 feet, respectively.

The effect of engine-inlet total temperature on exhaust-gas total temperature is presented in figure 4 for a constant actual engine speed of 7950 rpm. A decrease in inlet total temperature from 532° to 465° R resulted in a decrease in exhaust-gas total temperature of approximately 50° R, and any further decrease in inlet temperature caused the exhaust-gas temperature to increase. The data for the performance variables presented in figure 4 along with other engine performance data are included in table III.

The effect of engine-inlet ram-pressure ratio on corrected exhaust-gas total temperature and the corresponding net thrust variation for various corrected engine speeds are shown in figure 5. The decrease in corrected exhaust-gas total temperature as ram-pressure ratio is increased results from an increase in effective flow area of the exhaust nozzle, which corresponds to an increase in nozzle flow coefficient. The change in effective flow area is caused by the fact that the exhaust nozzle is not fully choked and by the existence of a
2625 boundary layer of subsonic flow around the sonic jet. This layer of subsonic flow decreases in depth as the engine-inlet ram-pressure ratio is increased, thus increasing the effective area of the nozzle and reducing the tail-pipe temperature. The effect of this flow-area change becomes constant after a ram-pressure ratio of approximately 1.25 (which corresponds to a tail-pipe pressure ratio of approximately 2.5) is attained. At this ram-pressure ratio of 1.25, the net thrust loss is approximately 3 percent of the thrust that could be obtained if the exhaust-gas total temperature had remained constant at the value obtained for an engine-inlet static condition. A tabulation of these data along with other engine performance parameters is given in table IV.

General Performance Calibration Data

The effect of Reynolds number index on generalized engine performance is shown in figures 6 to 10 where the corrected air flow, corrected fuel flow, corrected jet thrust parameter, corrected exhaust-gas total temperature, and engine pumping characteristics are presented. The variation of corrected air flow with corrected engine speed for various Reynolds number indices is presented in figure 6. At a corrected engine speed of 7950 rpm, the corrected air flow decreased from 104.0 to 99.2 pounds per second as Reynolds number index was decreased from 0.80 to 0.15. The corrected fuel flow (fig. 7) generalized for Reynolds number indices from 0.80 to 0.30 at corrected engine speeds above about 7500 rpm but increased with a further reduction of Reynolds number index. This increase in fuel flow results from the required rise in turbine-inlet temperature due to the decrease in compressor efficiency and the decrease in combustion efficiency at low Reynolds number indices. The increase in corrected fuel flow at rated corrected engine speed was approximately 8 percent as Reynolds number index was reduced from 0.30 to 0.15. The corrected jet thrust parameter, based on scale thrust readings, (fig. 8) generalized throughout the range of Reynolds number indices and corrected engine speeds investigated. Corrected exhaust-gas total temperature (fig. 9) generalized for Reynolds number indices from 0.80 to 0.30 but increased with a further reduction in Reynolds index. This increase in corrected exhaust-gas total temperature at the lower Reynolds numbers is attributed primarily to the decrease in compressor efficiency, which requires more work from the

turbine to maintain a given engine speed and hence a higher turbine-inlet temperature. Figure 10 illustrates the effect of Reynolds number index on the engine pumping characteristics. The relation between engine total-pressure ratio and engine total-temperature ratio is defined by a single line as Reynolds number index is decreased from 0.80 to 0.30 but shifts in the direction of increased engine total-temperature ratio at a given engine total-pressure ratio for a further reduction in Reynolds number index. This shift in the curves reflects the reduced efficiency of the compressor and turbine at conditions of 2625
low inlet Reynolds number.

The corrected engine windmilling speed is shown in figure 11 as a function of flight Mach number for altitudes from 15,000 to 45,000 feet. The corrected engine windmilling speed was unaffected by changes in altitude for the range of flight Mach numbers investigated.

The thrust is dependent upon the exhaust-gas temperature and in this investigation the gas temperatures were measured by the engine manufacturer's four-probe and five-probe thermocouple harnesses as well as the 25 NACA thermocouples. The readings of these different sets of instrumentation differ, with the result that the thrust at a given measured temperature will also vary. A comparison of the thrusts obtained is presented in the following table for NACA standard sea-level static conditions:

Performance based on	Engine speed (rpm)	Engine manufacturer's exhaust-gas thermocouple reading $T_{9,i}$ (°R)	Exhaust-gas total temperature based on NACA instrumentation T_9 (°R) (a)	Thrust (lb)
Exhaust-gas total temperature of 1710° R	7950	----	1710	5894
Engine manufacturer's five-probe thermocouple harness	7950	1710	1760	6074
Engine manufacturer's four-probe thermocouple harness	7950	1710	1766	6098

[a]Based on an average of 25 NACA thermocouples located 15.15 in. downstream of tail-cone-outlet flange.

The exhaust nozzle (area, 298.5 sq in.) was sized so as to give an exhaust-gas temperature of 1710° R (1250° F) at standard sea-level static conditions and rated engine speed. For this exhaust-gas temperature of 1710° R, the standard sea-level static thrust is 5894 pounds.

Because the engine is normally rated by the manufacturer for an exhaust-gas temperature based on a thermocouple reading obtained from the four- or five-probe thermocouple harness, thrust values have been included in the preceding table for the thermocouple reading of 1710° R obtained from the four- and five-probe systems with the corresponding gas temperatures included. The four- and five-probe harnesses indicated an exhaust-gas temperature between 50° and 60° lower than the true gas temperature and therefore give a correspondingly higher thrust for a given temperature limit based on a thermocouple reading. The method employed in calculating the thrust values is given in the appendix.

SUMMARY OF RESULTS

The following results were obtained from an investigation of the altitude performance of a J47-25 turbojet engine in an altitude chamber over a range of engine-inlet Reynolds number indices from 0.15 to 0.80:

1. At a constant engine speed, a decrease in inlet total temperature from 532° to 465° R resulted in a decrease in exhaust-gas total temperature of approximately 50° R.

2. At a given corrected engine speed and with critical pressure ratio existing in the exhaust nozzle, the corrected exhaust-gas temperature decreased as the ram-pressure ratio was increased from 1.0 to 1.25. Further increases in ram-pressure ratio had no effect on temperature. The corresponding net thrust loss at ram-pressure ratios of 1.25 and above, due to the reduction in exhaust-gas temperature below the limiting value, amounted to 3 percent.

3. At a corrected engine speed of 7950 rpm, the corrected air flow decreased from 104.0 to 99.2 pounds per second as Reynolds number index was decreased from 0.80 to 0.15.

4. Corrected exhaust-gas total temperature, corrected fuel flow, and engine pumping characteristics generalized for Reynolds number indices from 0.80 to 0.30 and the corrected jet thrust parameter generalized throughout the range of Reynolds number indices and corrected engine speeds investigated.

5. The corrected engine windmilling speed was unaffected by changes in altitude for the range of flight Mach numbers investigated.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 3, 1952

APPENDIX - METHODS OF CALCULATION

Symbols

The following symbols are used in the calculation and on the figures: 2625

A	area, sq ft
C_T	thermal expansion coefficient, ratio of hot exhaust-nozzle area to cold exhaust-nozzle area
C_d	ratio of effective flow area to physical flow area
C_j	jet thrust coefficient
F_d	thrust system scale reading, lb
F_j	jet thrust, lb
F_n	net thrust, lb
f/a	fuel-air ratio
g	acceleration due to gravity, 32.2 ft/sec^2
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft absolute
p	static pressure, lb/sq ft absolute
R	gas constant, 53.3 ft-lb/(lb)($^{\circ}$R)
Re	Reynolds number index, $\dfrac{\delta_1}{\varphi_1 \sqrt{\theta_1}}$
T	total temperature, $^{\circ}$R
T_i	indicated total temperature, $^{\circ}$R
V	velocity, ft/sec
W_a	air flow, lb/sec

W_f fuel flow, lb/hr

W_g gas flow, lb/sec

γ ratio of specific heats

δ ratio of engine-inlet total pressure P_1 to NACA standard sea-level pressure, 2116 lb/sq ft

2625

θ ratio of engine-inlet total temperature T_1 to NACA standard sea-level temperature, 519° R

φ ratio of coefficient of viscosity corresponding with T_1 to coefficient of viscosity corresponding with NACA standard sea-level temperature, 519° R

Subscripts:

0 free-stream conditions

0a bellmouth inlet

1 engine inlet

2 compressor inlet

3 compressor outlet

5 turbine outlet

9 exhaust-nozzle inlet

10 exhaust-nozzle outlet

cl compressor 12-stage leakage air flow

d thrust-cell measurement

e equivalent

i indicated

n vena contracta at exhaust-nozzle outlet

r rake

s scale

Calculations

Flight Mach number and velocity. - The flight Mach number assuming complete ram-pressure recovery was computed as

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1}\left[\left(\frac{P_1}{p_0}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]} \qquad (1)$$

2625

and

$$V_0 = M_0\sqrt{\gamma_1 g R T_1 \left(\frac{p_0}{P_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}}}$$

Temperature. - Total temperature was determined by a calibrated thermocouple with an impact-recovery factor of 0.85 from the indicated temperature by the following equation:

$$T = \frac{T_i\left(\frac{P}{p}\right)^{\frac{\gamma - 1}{\gamma}}}{1 + 0.85\left[\left(\frac{P}{p}\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \qquad (2)$$

Engine air flow. - Because of the large amount of air-flow leakage at the station where the engine inlet screens are mounted, the gas flow was determined at the exhaust-nozzle exit from total pressure and temperature at the nozzle inlet (station 9) by the following equation with the assumption that no energy loss occurred between the nozzle inlet and exit:

$$W_{g,n} = C_T\, C_d\, A_{10}\, p_n \sqrt{\frac{2\gamma_9}{\gamma_9 - 1}\, \frac{g}{RT_9}\left[\left(\frac{P_9}{p_n}\right)^{\frac{\gamma_9 - 1}{\gamma_9}} - 1\right]\left(\frac{P_9}{p_n}\right)^{\frac{\gamma_9 - 1}{\gamma_9}}} \qquad (3)$$

where in the subsonic case

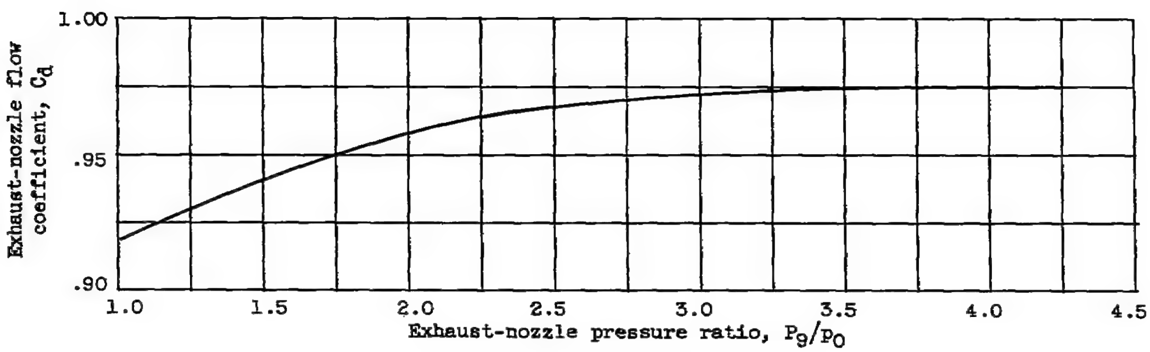
$$p_n = p_0$$

and in the choked case

$$p_n = \frac{P_9}{\left(\frac{1 + \gamma_9}{2}\right)^{\frac{\gamma_9}{\gamma_9 - 1}}}$$

2625

The value of the flow coefficient was determined from reference 2 using the area ratio and cone angle of the particular nozzle employed in this investigation. The magnitude of the flow coefficient is presented in the following curve:



The compressor-inlet air flow was then determined from the nozzle gas flow by

$$W_{a,2} = W_{g,n} - W_{f,e} + W_{a,cl} \tag{4}$$

where the compressor leakage air flow $W_{a,cl}$ was measured at two instrumented mid-frame bleed ports.

The engine-inlet air flow $W_{a,1}$ based on pressure and temperature measurements in a bellmouth mounted on the front of the engine was determined by the same general equation as for the tail-pipe gas flow. The percentage of leakage at the section housing the inlet screens is

$$W_{a,1-2} = \frac{W_{a,1} - W_{a,2}}{W_{a,2}}$$

and was 3.3 percent of the compressor-inlet air flow $W_{a,2}$ for the range of conditions covered in this investigation.

Thrusts. - The jet thrust as determined from the thrust system measurements was calculated from the equation

$$F_{j,s} = F_d + (A_{seal} - A_9)(P_1 - p_{seal}) + A_9(P_1 - p_0) + 0.80\left(\frac{1}{2}\frac{W_{a,1}}{g} V_{0a}\right) \tag{5}$$

2625

where the last term is the momentum force existing at the bellmouth inlet which was experimentally determined by instrumentation located on the surfaces of the bellmouth and bullet along with instrumentation at station 1. The net thrust will be determined by subtracting the equivalent momentum of the air at the engine inlet from the jet thrust.

$$F_{n,s} = F_{j,s} - \frac{W_{a,1} V_0}{g} = F_{j,s} - \frac{(W_{a,2} + W_{a,1-2})V_0}{g} \tag{6}$$

Jet thrust coefficient. - The jet thrust coefficient is defined as the ratio of scale jet thrust to rake jet thrust:

$$C_j = \frac{F_{j,s}}{F_{j,r}} \tag{7}$$

where

$$F_{j,r} = \frac{W_{g,n}}{g} V_n + A_n(p_n - p_0) \tag{8}$$

The charts in reference 3 were used in the solution of the preceding equation. When all the data obtained in this investigation were employed, the jet thrust coefficient was found to be independent of exhaust-nozzle pressure ratio and was a constant value of 0.99. The scatter in the coefficient values was approximately ±1 percent for the range of conditions investigated.

Determination of performance for particular flight condition. - For a given flight condition, values of Re, δ, and θ can be obtained from table II. If these generalizing parameter values and engine speed are known, air flow, fuel flow, and exhaust-gas temperature can be obtained from the various performance curves. In order to determine

the net thrust, the jet thrust parameter must first be corrected to the desired flight condition to obtain the jet thrust. Then in order to obtain net thrust, the leakage between stations 1 and 2 must be added to the air flow for station 2 so that

$$F_n = F_j - \left(\frac{W_{a,2} + W_{a,1-2}}{g}\right) V_0$$

2625

Sea-level static thrust ratings. - Because of the effect of inlet ram pressure on exhaust-gas temperature, data taken at an altitude of 5000 feet and flight Mach number of 0.2, which are included in the following table, had to be corrected to sea-level static conditions in order to determine the sea-level thrust for the engine.

Engine-inlet total pressure P_1 (lb/sq ft abs)	Engine-inlet total temperature T_1 (°R)	Nozzle-inlet total pressure P_9 (lb/sq ft abs)	Nozzle-inlet total temperature T_9 (°R)	Engine manufacturer's 4-probe nozzle-inlet indicated temperature $T_{9,i}$ (°R)	Engine manufacturer's 5-probe nozzle-inlet indicated temperature $T_{9,i}$ (°R)	Corrected engine speed $N/\sqrt{\theta_1}$ (rpm)	Corrected compressor-inlet air flow $W_{a,2}\sqrt{\theta_1}/\delta_1$ (lb/sec)	Corrected compressor leakage air flow $W_{c1}\sqrt{\theta_1}/\delta_1$ (lb/sec)	Corrected engine fuel flow $\frac{W_{f,2}}{\delta_1\sqrt{\theta_1}}$ (lb/hr)
1812	537	3050	1568	1522	1519	7281	95.7	1.9	4681
1814	537	3145	1612	1560	1560	7443	95.6	1.9	5008
1813	534	3154	1601	1556	1553	7464	98.6	2.0	5014
1812	537	3233	1656	1601	1604	7594	99.4	2.0	5348
1816	537	3370	1728	1674	1678	7813	101.8	2.0	5870
1813	537	3366	1731	1672	1680	7816	101.2	2.0	5916
1814	535	3397	1736	1679	1683	7846	102.1	2.0	6022

For sea-level static engine-inlet conditions, an engine speed of 7950 rpm, and a given exhaust-gas temperature, the tail-pipe total pressure may be determined from the engine-pumping-characteristic curves; therefore, the pressure ratio across the exhaust nozzle may also be determined. A plot of corrected fuel flow against engine temperature ratio will give the fuel flow for the proper exhaust-gas temperature. The compressor-inlet air flow may be determined from a plot of corrected air flow against corrected engine speed. In order to determine tail-pipe gas flow, compressor leakage air flow must be deducted and fuel flow added to the inlet air flow. From fuel flow, air flow, and exhaust-gas temperature, a value for γ_9 may be obtained. All the factors that are required to calculate the rake jet thrust from equation (8) are now known. To the rake jet thrust there must be applied a jet thrust coefficient obtained from the value presented in this appendix in order to obtain the final sea-level jet thrust value.

The preceding sea-level static thrust calcualtion required the use of two assumptions:

(1) The required nozzle-area change for the range of exhaust-gas temperatures of interest has no effect on the engine pumping characteristics.

(2) The required nozzle-area change for the small change in exhaust-gas temperature has no effect on the curve of corrected air flow against corrected engine speed. Both of these assumptions were checked with data that were obtained during this investigation and verified as accurate and logical assumptions.

REFERENCES

1. Walker, Curtis L., Huntley, S. C., and Braithwaite, W. M.: Component and Over-All Performance Evaluation of an Axial-Flow Turbojet Engine over a Range of Engine-Inlet Reynolds Numbers. NACA RM E52B08, 1952.

2. Grey, Ralph E., Jr., and Wilstead, H. Dean: Performance of Conical Jet Nozzles in Terms of Flow and Velocity Coefficients. NACA Rep. 933, 1949. (Supersedes NACA TN 1757.)

3. Turner, L. Richard, Addie, Albert N., and Zimmerman, Richard H.: Charts for the Analysis of One-Dimensional Steady Compressible Flow. NACA TN 1419, 1948.

2625

2625

TABLE I. - STANDARD

Reynolds number index Re	Engine speed N (rpm)	Altitude static pressure p_0 (lb/sq ft abs)	Engine-inlet total pressure P_1 (lb/sq ft abs)	Ram-pressure ratio P_1/p_0	Engine-inlet total temperature T_1 (°R)	Compressor-outlet total pressure P_3 (lb/sq ft abs)	Compressor-outlet total temperature T_3 (°R)	Turbine-inlet total pressure, P_4 (lb/sq ft abs)	Turbine-outlet total pressure, P_5 (lb/sq ft abs)	Turbine-outlet total temperature, T_5 (°R)	Nozzle-inlet total pressure, P_9 (lb/sq ft abs)	Nozzle-inlet static pressure, p_9 (lb/sq ft abs)	Nozzle-inlet total temperature, T_9 (°R)
0.147	5953	205	232	1.132	415	902	661	860	346	1128	333	263	1102
.151	6362	206	238	1.154	415	1058	695	1008	383	1234	376	292	1209
.156	7193	203	246	1.209	415	1335	761	1267	490	1481	468	364	1477
.154	7409	201	241	1.199	414	1358	780	1292	504	1570	481	372	1563
.151	7578	196	237	1.205	414	1365	800	1319	516	1662	492	381	1632
.150	7707	192	235	1.221	413	1412	810	1344	528	1721	504	390	1729
.202	5927	218	315	1.449	412	1211	650	1152	443	1035	427	327	1007
.200	6362	215	313	1.457	413	1403	684	1337	513	1164	491	378	1162
.201	6801	212	313	1.475	412	1582	718	1501	577	1296	555	427	1300
.202	7205	211	316	1.497	412	1713	753	1626	631	1442	604	466	1442
.200	7407	209	313	1.499	412	1780	772	1692	655	1521	627	484	1531
.199	7574	206	311	1.513	413	1829	791	1743	675	1606	646	501	1602
.198	7726	210	311	1.482	414	1899	807	1807	702	1686	671	521	1684
.249	5921	241	392	1.622	414	1483	647	1409	543	998	523	400	992
.248	6358	238	389	1.634	414	1724	682	1644	632	1130	608	468	1138
.248	6815	237	389	1.642	414	1958	718	1860	714	1270	686	527	1279
.250	7207	238	391	1.642	413	2118	750	2010	777	1420	745	576	1418
.247	7409	238	389	1.632	414	2189	770	2081	805	1490	772	597	1487
.249	7570	237	389	1.639	413	2276	787	2167	837	1564	802	619	1571
.248	7818	239	388	1.619	413	2349	814	2238	871	1686	834	648	1707
.299	5932	367	470	1.280	414	1817	652	1727	675	1006	650	506	1011
.296	6358	354	468	1.323	416	2091	684	1994	764	1117	735	566	1128
.302	6360	273	473	1.732	413	2092	679	1995	764	1111	735	567	1122
.296	6815	351	468	1.333	416	2335	719	2218	854	1253	819	631	1271
.311	6822	280	486	1.738	412	2446	715	2324	890	1248	855	659	1268
.303	7193	277	473	1.711	412	2561	746	2431	936	1379	899	698	1398
.297	7195	344	468	1.360	415	2523	752	2395	926	1388	888	688	1400
.298	7407	338	466	1.377	415	2622	770	2494	965	1480	926	716	1484
.300	7415	278	469	1.686	412	2651	766	2521	974	1471	934	724	1488
.297	7570	338	466	1.380	413	2699	786	2571	995	1542	954	737	1557
.303	7574	276	473	1.712	412	2755	784	2624	1024	1553	981	759	1570
.302	7725	279	469	1.681	411	2818	800	2685	1054	1630	1009	782	1652
.401	5923	553	740	1.338	467	2481	699	2345	918	998	886	702	1003
.404	6360	555	745	1.339	466	2919	733	2775	1072	1116	1030	800	1135
.401	6813	555	744	1.340	469	3349	774	3194	1233	1263	1182	911	1294
.405	7199	547	739	1.351	464	3699	800	3516	1356	1381	1302	1006	1413
.397	7405	558	737	1.320	470	3834	824	3638	1403	1459	1347	1043	1495
.404	7570	554	747	1.349	468	3973	840	3769	1468	1521	1410	1090	1555
.399	7947	549	741	1.349	470	4258	878	4056	1580	1706	1521	1176	1732
.406	7956	552	746	1.350	466	4285	874	4089	1580	1706	1523	1184	1748
.434	5930	506	718	1.418	430	2619	663	2483	958	969	922	719	979
.431	6362	508	721	1.420	434	3013	703	2869	1108	1100	1062	819	1116
.419	6817	505	719	1.425	443	3371	746	3216	1240	1242	1169	918	1268
.431	7112	507	725	1.429	436	3728	763	3539	1357	1336	1302	1006	1369
.425	7407	509	720	1.416	438	3925	792	3728	1435	1444	1380	1065	1475
.419	7565	510	720	1.412	443	3988	812	3801	1469	1512	1415	1095	1541
.428	7741	511	728	1.424	441	4137	830	3937	1524	1591	1468	1138	1622
.419	7962	511	710	1.391	438	4251	849	4052	1585	1702	1527	1189	1735
.510	5929	482	940	1.951	467	3085	696	2907	1106	928	1080	816	940
.509	6362	478	944	1.977	469	3629	735	3442	1317	1081	1262	972	1106
.509	6817	480	939	1.956	467	4201	770	4006	1537	1234	1475	1137	1271
.508	7193	482	940	1.951	468	4609	801	4387	1689	1347	1624	1252	1399
.508	7409	474	939	1.983	468	4750	818	4504	1776	1417	1699	1312	1469
.508	7566	481	942	1.961	469	4985	835	4730	1829	1489	1759	1363	1537
.502	7718	477	931	1.952	469	5073	851	4818	1882	1557	1784	1391	1602
.509	7722	485	939	1.936	467	5138	848	4880	1887	1551	1818	1408	1600
.503	7943	481	930	1.935	468	5268	872	5011	1948	1666	1873	1457	1711
.505	7953	482	940	1.951	471	5320	875	5062	1960	1666	1891	1467	1711
.610	5930	520	1127	2.166	468	3663	697	3442	1311	916	1256	966	931
.613	6362	519	1121	2.161	464	4352	730	4128	1582	1065	1514	1166	1097
.610	6815	520	1123	2.159	467	5008	768	4773	1831	1220	1759	1356	1258
.605	7195	526	1124	2.138	470	5492	802	5229	2016	1345	1955	1498	1393
.608	7411	520	1118	2.148	466	5764	816	5470	2107	1511	2031	1571	1466
.609	7578	524	1123	2.142	467	5981	831	5675	2190	1477	2111	1635	1526
.602	7722	528	1125	2.140	472	6122	852	5812	2245	1543	2167	1677	1591
.607	7949	530	1131	2.134	471	6419	874	6107	2369	1648	2288	1775	1708
.596	7951	520	1114	2.144	472	6302	875	5994	2320	1655	2240	1740	1712
.809	5929	1060	1789	1.688	537	4913	760	4608	1788	924	1729	1368	934
.807	6372	1050	1788	1.703	538	5884	801	5567	2134	1076	2047	1587	1086
.807	6817	1050	1785	1.700	537	6894	841	6562	2523	1234	2419	1865	1268
.809	7197	1050	1790	1.705	537	7732	874	7392	2815	1377	2735	2111	1419
.809	7409	1047	1789	1.709	537	8153	892	7807	3013	1450	2897	2241	1498
.807	7568	1050	1786	1.701	537	8449	905	8098	3126	1497	3008	2331	1549
.809	7727	1053	1788	1.698	537	8763	918	8389	3249	1566	3123	2421	1612
.803	7951	1034	1779	1.721	538	9155	939	8744	3385	1669	3261	2535	1701

NACA

2625

ENGINE PERFORMANCE DATA

Air frame mfg. 4-probe nozzle-inlet total pressure, P_9 (lb/sq ft abs)	Air frame mfg. 3-probe nozzle-inlet total pressure, P_9 (lb/sq ft abs)	Engine mfg. 4-probe nozzle-inlet indicated temperature $T_{9,i}$, (°R)	Engine mfg. 5-probe nozzle-inlet indicated temperature, $T_{9,i}$ (°R)	Compressor-inlet air flow, $W_{a,2}$ (lb/sec)	Engine fuel flow, $W_{f,e}$ (lb/hr)	Fuel-air ratio f/a	Jet thrust F_j (lb)	Net thrust F_n (lb)	Corrected engine speed $N/\sqrt{\theta_1}$ (rpm)	Corrected compressor-inlet air flow $W_{a,2}\sqrt{\theta_1}/\delta_1$ (lb/sec)	Corrected engine fuel flow $W_{f,e}/\delta_1\sqrt{\theta_1}$ (lb/hr)	Corrected exhaust-gas total temperature, T_9/θ_1 (°R)	Corrected jet thrust parameter $(F_j+p_0A_n)/\delta_1$ (lb)
340	336	1057	1058	10.3	374	0.0103	391	258	6655	83.7	3815	1379	7450
382	382	1188	1197	11.4	462	.0115	519	357	7113	90.5	4598	1512	8420
471	477	1427	1446	12.9	696	.0153	753	542	8042	99.4	6696	1848	10,950
482	489	1512	1521	12.9	762	.0168	801	596	8298	101.2	7484	1960	10,690
493	498	1582	1590	12.9	822	.0181	817	611	8487	103.1	8233	2055	10,110
505	508	1668	1670	12.9	883	.0195	888	677	8640	103.5	8931	2173	11,600
437	432	981	986	14.2	438	.0087	603	281	6650	85.1	3299	1269	7080
498	499	1124	1132	15.3	575	.0106	791	437	7132	92.4	4357	1461	8360
560	566	1265	1274	16.4	730	.0126	968	588	7631	98.9	5532	1638	9520
607	615	1395	1405	17.0	883	.0147	1105	704	8084	101.6	6641	1817	10,340
628	637	1471	1478	17.1	973	.0161	1165	763	8311	103.1	7388	1929	10,810
647	653	1551	1558	17.2	1059	.0175	1227	819	8490	104.4	8067	2014	11,240
672	676	1636	1639	17.4	1156	.0189	1262	862	8652	105.6	8798	2124	11,540
535	529	964	967	17.7	516	.0082	829	371	6632	85.3	3124	1244	7190
615	617	1105	1111	19.3	687	.0101	1049	561	7121	93.6	4186	1427	8390
692	699	1240	1244	20.5	883	.0122	1246	713	7633	99.5	5374	1604	9440
747	758	1369	1375	21.1	1062	.0142	1398	848	8079	102.1	6448	1782	10,250
772	783	1446	1455	21.3	1161	.0154	1459	919	8298	103.8	7083	1877	10,690
801	813	1513	1524	21.6	1255	.0165	1544	985	8486	105.0	7653	1975	11,080
832	841	1640	1648	21.6	1402	.0185	1629	1078	8784	105.2	8583	2146	11,610
661	661	974	982	21.4	628	.0083	849	445	6644	86.1	3167	1268	7250
744	748	1097	1104	23.2	820	.0100	1087	628	7102	93.8	4142	1408	8240
743	747	1088	1094	23.5	814	.0098	1281	639	7130	93.6	4082	1410	8270
826	835	1233	1239	24.4	1038	.0120	1372	873	7612	98.9	5243	1586	9500
861	871	1227	1233	25.7	1074	.0118	1578	871	7654	99.8	5247	1598	9400
901	914	1358	1360	25.7	1285	.0139	1736	1044	8071	102.6	6546	1761	10,330
890	902	1359	1362	25.3	1246	.0140	1539	1011	8044	102.2	6294	1751	10,180
925	939	1438	1444	25.6	1376	.0152	1670	1122	8303	103.9	7008	1865	10,780
935	948	1437	1447	25.9	1388	.0152	1830	1141	8320	104.3	7053	1872	10,870
954	966	1508	1514	25.8	1463	.0161	1720	1175	8486	104.6	[illegible]	1957	11,000
981	993	1519	1523	26.4	1524	.0163	1960	1253	8498	105.3	[illegible]	1978	11,340
1008	1018	1597	1602	26.6	1647	.0176	1973	1278	8683	106.7	8351	2086	11,510
895	901	969	977	28.7	786	.0075	1067	459	6243	77.8	2309	1114	6330
1037	1049	1102	1107	32.2	1082	.0085	1469	766	6710	87.1	3193	1264	7460
1188	1197	1256	1258	34.9	1395	.0114	1858	1096	7167	94.3	4172	1432	8560
1305	1315	1371	1372	36.9	1708	.0130	2185	1374	7617	99.8	5172	1581	9500
1348	1358	1450	1448	37.1	1872	.0144	2248	1465	7783	[illegible]	5847	1650	9780
1409	1419	1508	1505	38.1	2049	.0153	2447	1610	7971	102.5	8115	1724	10,190
1518	1522	1679	1678	38.9	2480	.0182	2705	1862	8352	105.8	7444	1912	10,980
1523	1526	1698	1691	38.8	2490	.0182	2743	1895	8394	104.4	[illegible]	1947	11,030
933	941	950	957	31.0	838	.0076	1276	568	6517	[illegible]	2716	[illegible]	6860
1066	1084	1084	1086	33.7	1116	.0094	1631	856	6960	90.4	3582	[illegible]	7880
1191	1208	1231	1231	35.6	1418	.0113	1992	1184	7383	96.6	4516	1486	8940
1303	1318	1331	1329	37.6	1700	.0128	2279	1412	7759	100.5	5418	1629	9730
1378	1394	1431	1431	38.3	1944	.0144	2478	1610	8066	103.4	6219	1748	10,380
1414	1427	1496	1498	38.5	2082	.0154	2557	1689	8193	104.4	6631	1806	10,630
1464	1479	1570	1575	38.9	2268	.0166	2697	1798	8399	104.3	7153	1909	10,920
1522	1533	1678	1682	39.1	2555	.0186	2809	1959	8671	107.0	8294	[illegible]	11,530
1075	1083	911	917	37.0	858	.0066	1702	508	6249	79.0	2036	1044	6080
1272	1286	1079	1080	40.8	1218	.0087	2231	898	6893	88.8	[illegible]	1224	7220
1478	1495	1240	1236	44.5	1708	.0109	2779	1341	7185	95.2	4056	1412	8510
1625	1640	1362	1356	46.8	2098	.0127	3169	1675	7574	100.0	4975	1551	9386
1699	1714	1424	1422	47.7	2317	.0138	3385	1842	7802	102.1	5497	1629	9840
1758	1772	1491	1490	48.3	2513	.0148	3540	1987	7959	103.0	5935	1701	10,190
1793	1805	1555	1555	48.2	2681	.0158	3618	2081	8119	[illegible]	6411	1775	10,470
1815	1828	1550	1551	48.9	2722	.0158	3644	2096	8139	104.6	6467	1778	10,480
1871	1881	1659	1661	48.6	3015	.0177	3819	2273	8364	105.1	[illegible]	1897	10,960
1889	1897	1659	1660	49.1	3032	.0176	3861	2289	8351	105.3	7163	1886	10,940
1275	1283	901	907	44.2	982	.0063	2119	577	6214	78.8	[illegible]	[illegible]	6010
1525	1544	1069	1069	49.2	1455	.0084	2796	1103	6725	87.9	[illegible]	1226	7510
1760	1783	1231	1227	53.5	2009	.0107	3466	1642	7188	95.5	3993	1399	8560
1939	1956	1357	1353	56.5	2480	.0125	3901	1994	7560	[illegible]	4908	1539	9400
2028	2042	1424	1422	57.2	2765	.0138	4127	2187	7819	102.6	5522	1633	9860
2112	2125	1483	1481	58.2	3005	.0147	4349	2382	7987	104.0	5967	1694	10,240
2164	2171	1546	1544	58.4	3217	.0157	4472	2497	8100	104.8	6348	1753	10,460
2280	2290	1659	1656	59.5	3656	.0175	4775	2765	8346	106.1	[illegible]	1882	10,980
2238	2243	1663	1659	58.2	3593	.0176	4697	2726	8341	105.4	7157	1883	10,970
1748	1755	899	907	58.7	1102	.0054	2180	328	5829	70.6	1282	905	5180
2057	2078	1051	1058	66.1	1716	.0074	3080	1010	6259	79.6	[illegible]	1067	6220
2409	2443	1230	1228	72.6	2538	.0100	4016	1748	6702	87.5	2957	1224	7340
2723	2750	1380	1373	77.9	3323	.0121	4839	2401	7075	93.7	3861	1371	8290
2882	2903	1454	1448	80.4	3785	.0134	5289	2794	7284	96.7	4379	1448	8830
2994	3007	1510	1504	82.1	4086	.0143	5539	3007	7410	98.9	4760	1497	9150
3115	3117	1565	1562	83.6	4406	.0150	5878	3305	7596	100.5	5124	1558	9540
3262	3256	1648	1652	84.9	4953	.0167	6252	3614	7809	102.8	5784	1641	9980

NACA

2625

TABLE II.- REYNOLDS NUMBER INDEX VARIATION WITH

FLIGHT MACH NUMBER AND ALTITUDE

[Ram-pressure recovery, 1.00.]

Altitude (ft)	Flight Mach number M_0	δ	θ	φ	Reynolds number index $\delta/\varphi\sqrt{\theta}$
0	0	1.000	1.000	1.000	1.000
	.1	1.007	1.002	1.002	1.004
	.2	1.028	1.008	1.006	1.018
	.3	1.064	1.018	1.013	1.041
	.4	1.117	1.032	1.023	1.075
	.5	1.186	1.050	1.036	1.117
	.6	1.276	1.072	1.051	1.173
	.7	1.387	1.098	1.069	1.238
	.8	1.524	1.128	1.090	1.316
	.9	1.691	1.162	1.117	1.404
	1.0	1.893	1.200	1.141	1.516
5,000	0	0.8318	0.9657	0.9753	0.8679
	.1	.8374	.9676	.9764	.8718
	.2	.8554	.9734	.9809	.8839
	.3	.8862	.9830	.9875	.9041
	.4	.9291	.9965	.9973	.9333
	.5	.9868	1.014	1.010	.9703
	.6	1.061	1.035	1.026	1.018
	.7	1.154	1.060	1.044	1.073
	.8	1.268	1.089	1.064	1.141
	.9	1.407	1.122	1.086	1.223
	1.0	1.575	1.159	1.117	1.309
10,000	0	0.6881	0.9312	0.9491	0.7513
	.1	.6923	.9331	.9504	.7541
	.2	.7075	.9387	.9549	.7647
	.3	.7320	.9480	.9621	.7814
	.4	.7684	.9609	.9714	.8069
	.5	.8157	.9776	.9836	.8388
	.6	.8776	.9983	.9989	.8794
	.7	.9542	1.022	1.016	.9291
	.8	1.048	1.050	1.037	.9859
	.9	1.163	1.082	1.058	1.057
	1.0	1.302	1.117	1.083	1.137
15,000	0	0.5643	0.8969	0.9223	0.6461
	.1	.5681	.8987	.9233	.6490
	.2	.5799	.9040	.9281	.6572
	.3	.6002	.9131	.9347	.6720
	.4	.6300	.9256	.9448	.6931
	.5	.6692	.9416	.9570	.7206
	.6	.7198	.9615	.9719	.7553
	.7	.7826	.9848	.9891	.7973
	.8	.8601	1.012	1.008	.8482
	.9	.9542	1.042	1.031	.9062
	1.0	1.068	1.076	1.055	.9762
20,000	0	0.4596	0.8626	0.8960	0.5523
	.1	.4629	.8644	.8966	.5553
	.2	.4726	.8696	.9016	.5622
	.3	.4891	.8780	.9072	.5754
	.4	.5132	.8902	.9172	.5930
	.5	.5454	.9058	.9289	.6170
	.6	.5865	.9247	.9440	.6461
	.7	.6375	.9470	.9610	.6817
	.8	.7004	.9728	.9798	.7248
	.9	.7769	1.002	1.002	.7746
	1.0	.8700	1.035	1.026	.8341
25,000	0	0.3710	0.8281	0.8682	0.4696
	.1	.3737	.8299	.8700	.4715
	.2	.3814	.8347	.8740	.4776
	.3	.3948	.8430	.8804	.4884
	.4	.4145	.8545	.8891	.5043
	.5	.4399	.8696	.9016	.5233
	.6	.4731	.8877	.9151	.5487
	.7	.5147	.9092	.9316	.5794
	.8	.5657	.9339	.9515	.6152
	.9	.6276	.9620	.9724	.6581
	1.0	.7023	.9934	.9950	.7082
30,000	0	0.2968	0.7938	0.8414	0.3959
	.1	.2989	.7954	.8430	.3975
	.2	.3052	.8002	.8469	.4029
	.3	.3158	.8081	.8525	.4121
	.4	.3315	.8193	.8621	.4248
	.5	.3519	.8335	.8727	.4416
30,000	0.6	0.3787	0.8509	0.8862	0.4633
	.7	.4118	.8715	.9029	.4886
	.8	.4522	.8954	.9207	.5190
	.9	.5019	.9222	.9416	.5551
	1.0	.5619	.9524	.9655	.5964
35,000	0	0.2352	0.7595	0.8149	0.3312
	.1	.2368	.7611	.8164	.3325
	.2	.2418	.7655	.8196	.3372
	.3	.2502	.7732	.8257	.3446
	.4	.2627	.7838	.8337	.3559
	.5	.2789	.7975	.8443	.3699
	.6	.3001	.8141	.8576	.3878
	.7	.3262	.8339	.8727	.4093
	.8	.3583	.8566	.8910	.4346
	.9	.3977	.8825	.9111	.4647
	1.0	.4452	.9112	.9334	.4997
40,000	0	0.1853	0.7572	0.8130	0.2619
	.1	.1866	.7588	.8141	.2631
	.2	.1905	.7632	.8175	.2667
	.3	.1972	.7709	.8239	.2726
	.4	.2070	.7815	.8321	.2814
	.5	.2198	.7950	.8430	.2924
	.6	.2364	.8118	.8562	.3065
	.7	.2570	.8314	.8714	.3235
	.8	.2824	.8539	.8889	.3438
	.9	.3134	.8798	.9090	.3676
	1.0	.3506	.9085	.9310	.3951
45,000	0	0.1459	0.7572	0.8130	0.2062
	.1	.1469	.7588	.8141	.2071
	.2	.1500	.7632	.8175	.2100
	.3	.1552	.7709	.8239	.2145
	.4	.1630	.7815	.8321	.2216
	.5	.1730	.7950	.8430	.2302
	.6	.1862	.8118	.8562	.2414
	.7	.2024	.8314	.8714	.2548
	.8	.2224	.8539	.8889	.2708
	.9	.2467	.8798	.9090	.2894
	1.0	.2762	.9085	.9310	.3112
50,000	0	0.1149	0.7572	0.8130	0.1624
	.1	.1157	.7588	.8141	.1631
	.2	.1181	.7632	.8175	.1654
	.3	.1223	.7709	.8239	.1691
	.4	.1284	.7815	.8321	.1746
	.5	.1362	.7950	.8430	.1812
	.6	.1466	.8118	.8562	.1900
	.7	.1594	.8314	.8714	.2006
	.8	.1751	.8539	.8889	.2132
	.9	.1943	.8798	.9090	.2279
	1.0	.2175	.9085	.9310	.2451
55,000	0	0.0905	0.7572	0.8130	0.1279
	.1	.0911	.7588	.8141	.1285
	.2	.0930	.7632	.8175	.1302
	.3	.0963	.7709	.8239	.1331
	.4	.1011	.7815	.8321	.1374
	.5	.1073	.7950	.8430	.1428
	.6	.1155	.8118	.8562	.1497
	.7	.1255	.8314	.8714	.1580
	.8	.1379	.8539	.8889	.1679
	.9	.1530	.8798	.9090	.1795
	1.0	.1713	.9085	.9310	.1930
60,000	0	0.0713	0.7572	0.8130	0.1008
	.1	.0717	.7588	.8141	.1011
	.2	.0733	.7632	.8175	.1026
	.3	.0758	.7709	.8239	.1048
	.4	.0796	.7815	.8321	.1082
	.5	.0845	.7950	.8430	.1124
	.6	.0909	.8118	.8562	.1178
	.7	.0988	.8314	.8714	.1244
	.8	.1086	.8539	.8889	.1322
	.9	.1205	.8798	.9090	.1413
	1.0	.1349	.9085	.9310	.1520

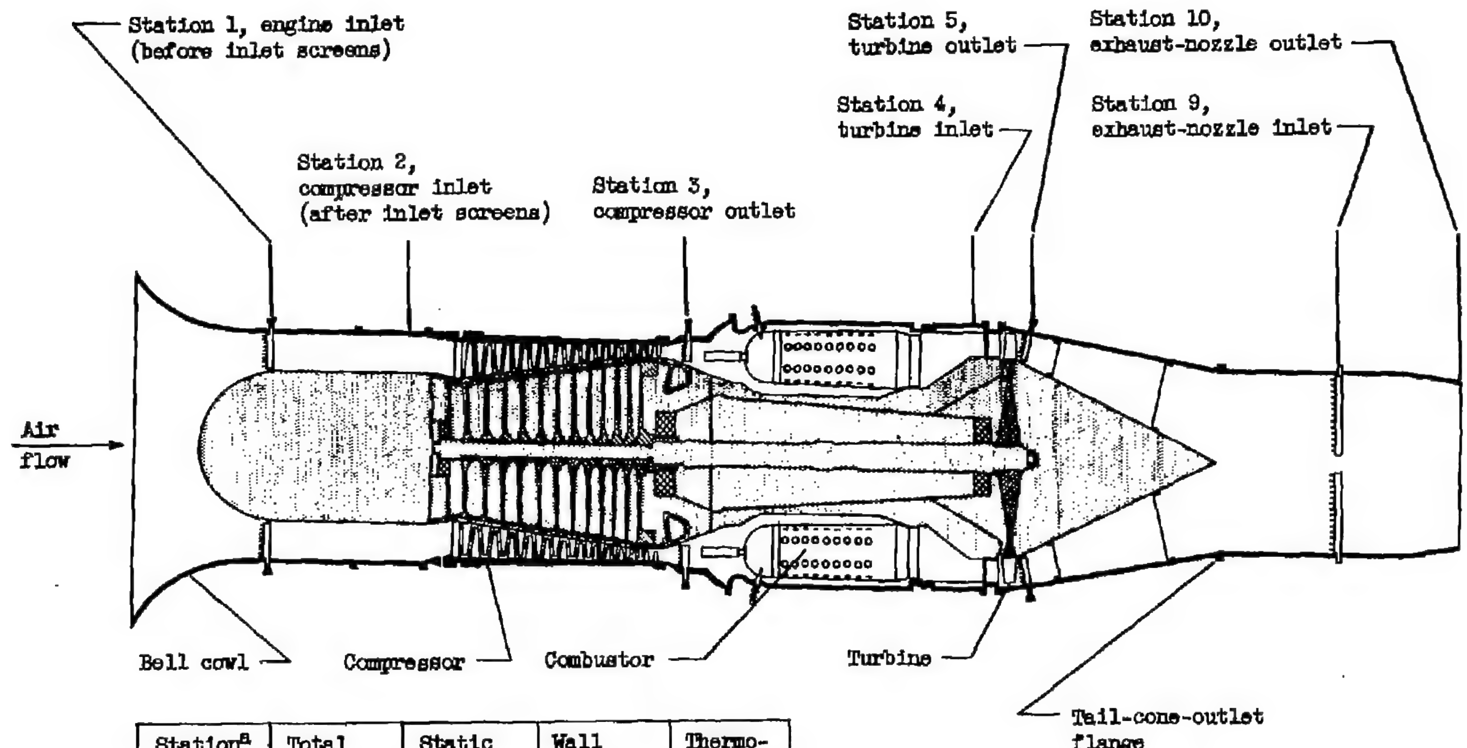
TABLE III. - PERFORMANCE DATA FOR EFFECT OF ENGINE-INLET TOTAL TEMPERATURE ON EXHAUST-GAS TOTAL TEMPERATURE

Engine speed N (rpm)	Altitude static pressure p_0 (lb/sq ft abs)	Engine-inlet total pressure P_1 (lb/sq ft abs)	Engine-inlet static pressure p_1 (lb/sq ft abs)	Engine-inlet total temperature T_1 (°R)	Nozzle-inlet total pressure P_9 (lb/sq ft abs)	Nozzle-inlet total temperature T_9 (°R)	Compressor-inlet air flow $W_{a,2}$ (lb/sec)	Engine fuel flow $W_{f,e}$ (lb/hr)	Net thrust F_n (lb)	Corrected engine speed $N/\sqrt{\theta_1}$ (rpm)	Corrected exhaust-gas total temperature T_9/θ_1 (°R)
7947	987	996	894	431	2102	1690	54.9	3485	3063	8718	2035
7947	970	1002	898	455	2027	1678	53.1	3260	2881	8487	1915
7947	972	999	901	481	1955	1681	51.1	3084	2696	8257	1814
7951	986	999	902	499	1908	1897	49.6	2979	2572	8110	1765
7953	969	991	895	520	1869	1713	48.3	2911	2490	7945	1710
7953	969	995	900	532	1853	1731	47.7	2875	2422	7855	1689

TABLE IV. - PERFORMANCE DATA FOR EFFECT OF ENGINE-INLET RAM-PRESSURE RATIO ON CORRECTED EXHAUST-GAS TOTAL TEMPERATURE

Engine speed N (rpm)	Altitude static pressure p_0 (lb/sq ft abs)	Engine-inlet total pressure P_1 (lb/sq ft abs)	Engine-inlet static pressure p_1 (lb/sq ft abs)	Engine-inlet total temperature T_1 (°R)	Nozzle-inlet total pressure P_9 (lb/sq ft abs)	Nozzle-inlet total temperature T_9 (°R)	Compressor-inlet air flow $W_{a,2}$ (lb/sec)	Engine fuel flow $W_{f,e}$ (lb/hr)	Net thrust F_n (lb)	Corrected engine speed $N/\sqrt{\theta_1}$ (rpm)	Corrected exhaust-gas total temperature T_9/θ_1 (°R)
7953	1294	1332	1204	512	2560	1741	65.3	4022	3460	8009	1765
7951	1223	1335	1207	513	2545	1731	65.3	3975	3229	7999	1752
7955	1173	1339	1211	512	2544	1729	65.4	3973	3176	8011	1753
7945	1125	1340	1211	511	2540	1719	65.7	3948	3092	8009	1747
7947	1042	1342	1213	512	2545	1718	66.0	3948	3040	8003	1742
7951	1290	1331	1207	529	2497	1758	63.3	3902	3307	7876	1725
7947	1220	1336	1212	528	2493	1743	63.6	3874	3077	7879	1713
7953	1169	1337	1211	529	2483	1741	63.5	3829	3010	7877	1708
7947	1120	1340	1214	529	2484	1738	63.7	3865	2985	7872	1705
7951	1083	1333	1207	529	2478	1732	63.8	3865	2922	7875	1699
7943	1047	1340	1212	529	2479	1728	64.0	3856	2888	7868	1695
7945	1455	1533	1394	536	2842	1743	71.7	4330	3562	7818	1688
7951	1464	1614	1467	537	2981	1727	75.7	4502	3569	7817	1669
7953	1471	1762	1597	536	3256	1726	83.1	4933	3790	7826	1671
7951	1468	1896	1713	537	3486	1712	89.7	5300	3987	7817	1655
7720	1764	1818	1659	531	3304	1669	84.0	4779	4228	7632	1631
7737	1726	1815	1655	536	3273	1673	82.2	4710	4086	7613	1620
7722	1691	1816	1658	535	3250	1657	82.0	4638	4001	7605	1607
7727	1597	1819	1658	534	3247	1653	82.3	4628	3922	7617	1607
7724	1516	1824	1659	534	3247	1642	82.8	4618	3832	7614	1596
7727	1441	1826	1659	533	3256	1642	83.3	4648	3800	7625	1599

NACA

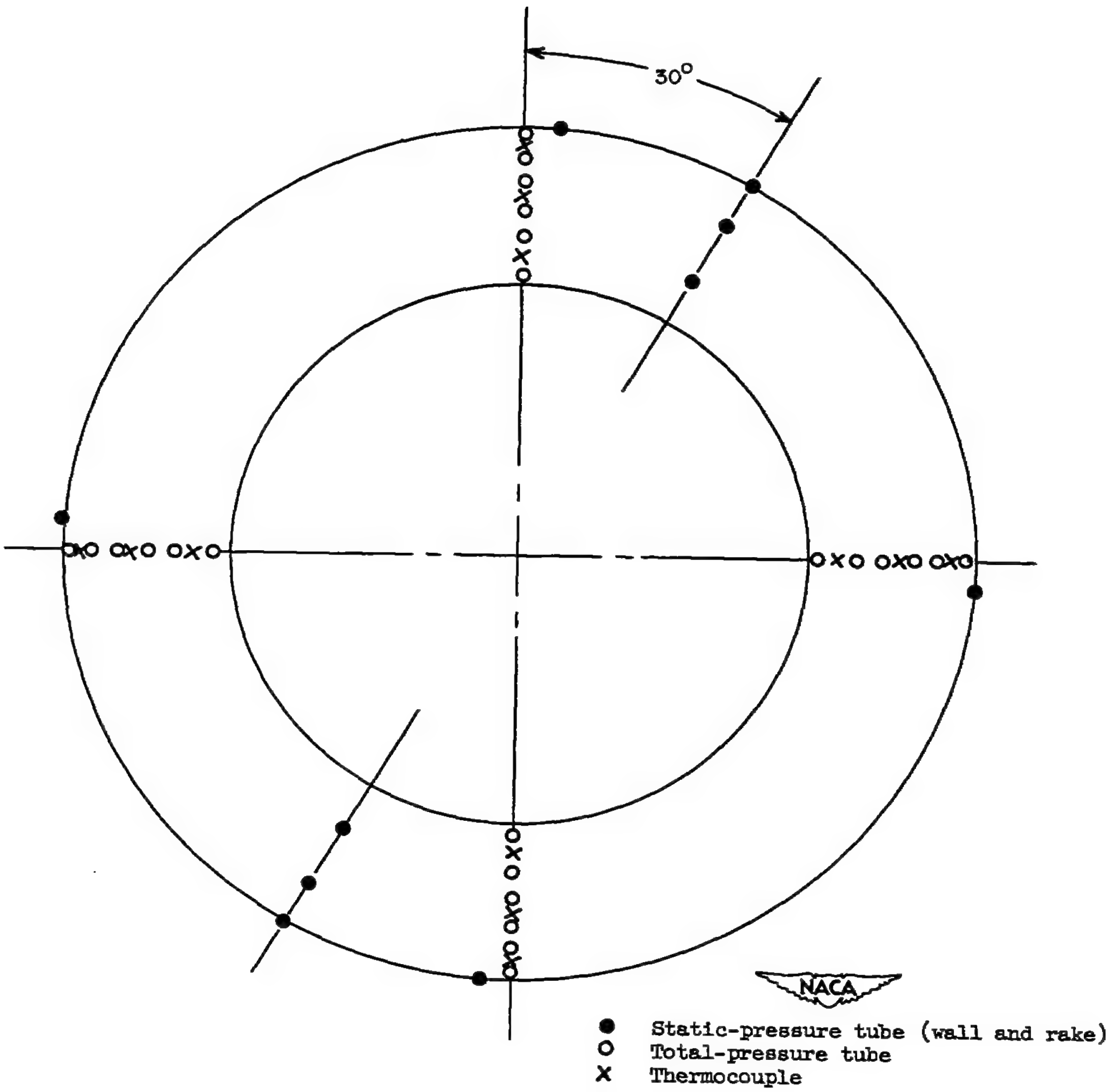


Station[a]	Total pressure tubes	Static pressure tubes	Wall static orifices	Thermo-couples
1	24	4	6	12
3	12	0	2	12
4	8	0	0	8
5	15	0	3	12
9	25	0	5	41

[a]No instrumentation at station 2.

Figure 1. - Cross section of engine showing location of instrumentation.

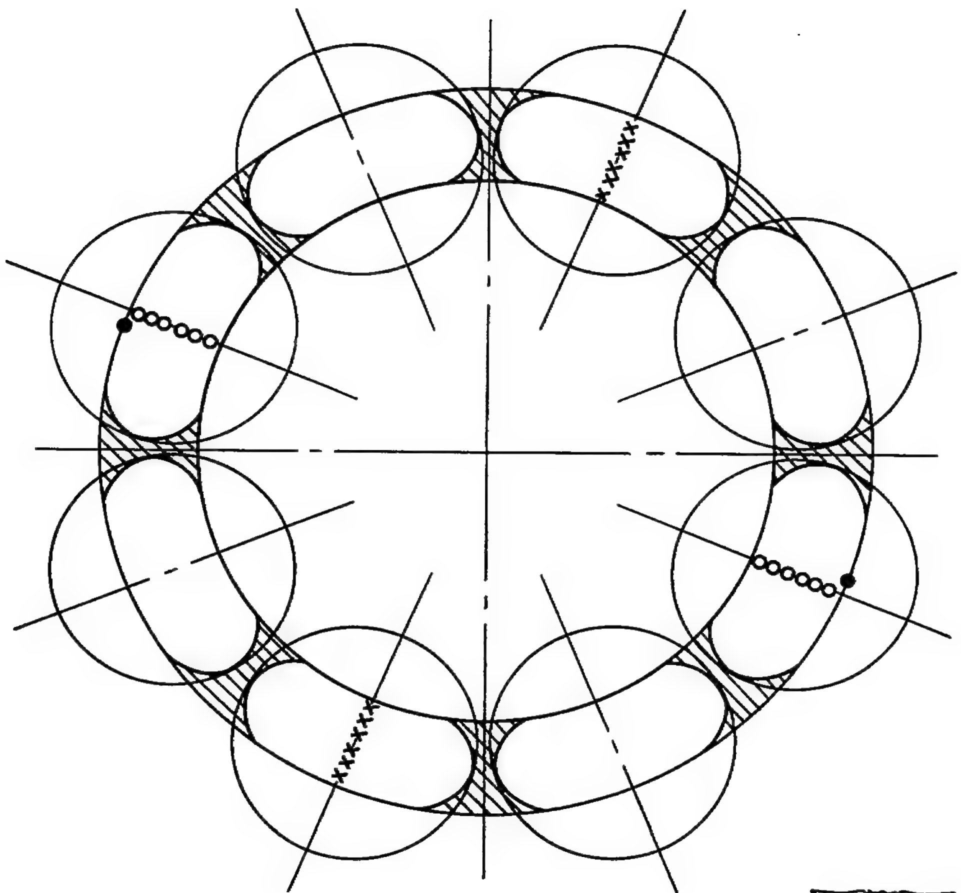
2625



(a) Instrumentation at engine inlet, station 1, 21 inches upstream of leading edge of compressor-inlet guide vanes. Viewed from upstream.

Figure 2. - Instrumentation sketches of various measuring stations.

2625

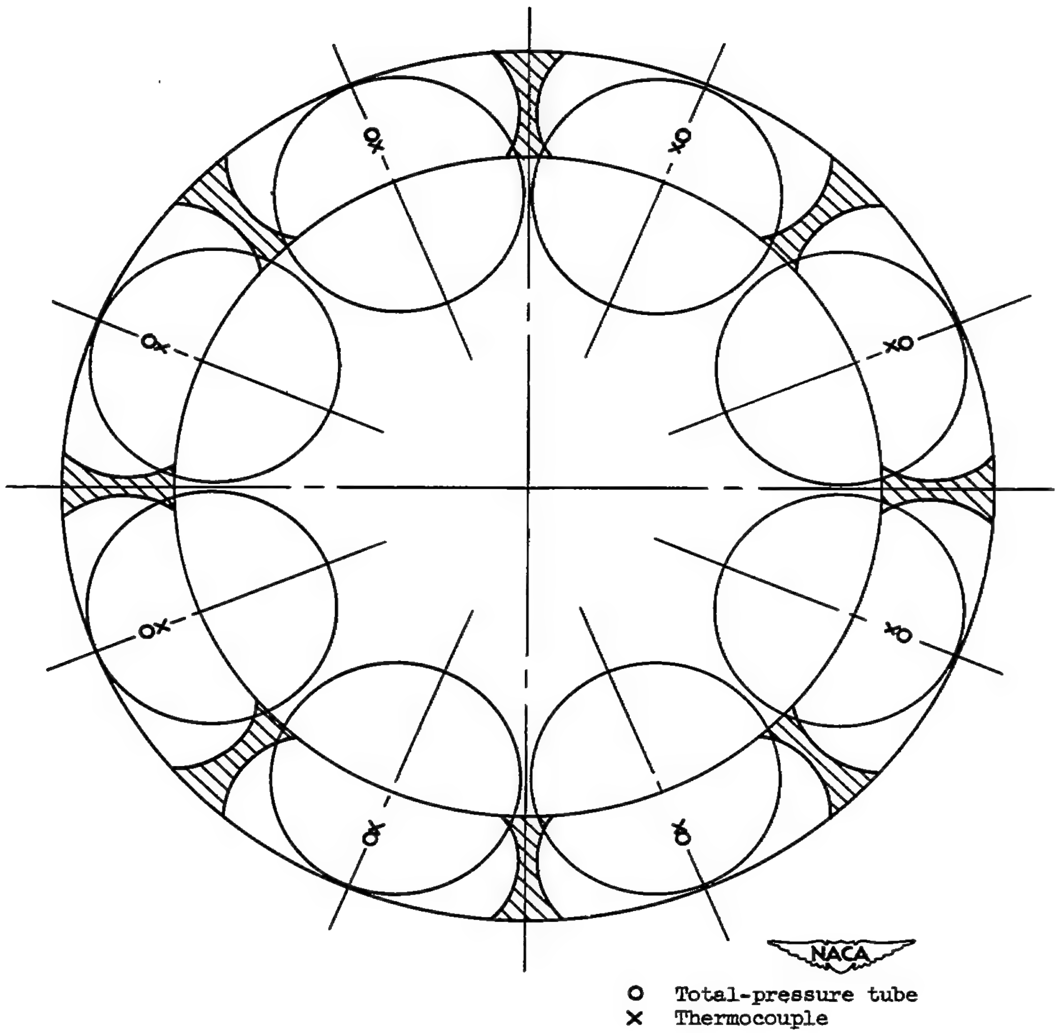


● Wall static-pressure tube
o Total-pressure tube
× Thermocouple

(b) Instrumentation at compressor outlet~~,~~ station 3, 2 inches downstream of trailing edge of compressor-outlet guide vanes. Viewed from upstream.

Figure 2. - Continued. Instrumentation sketches of various measuring stations.

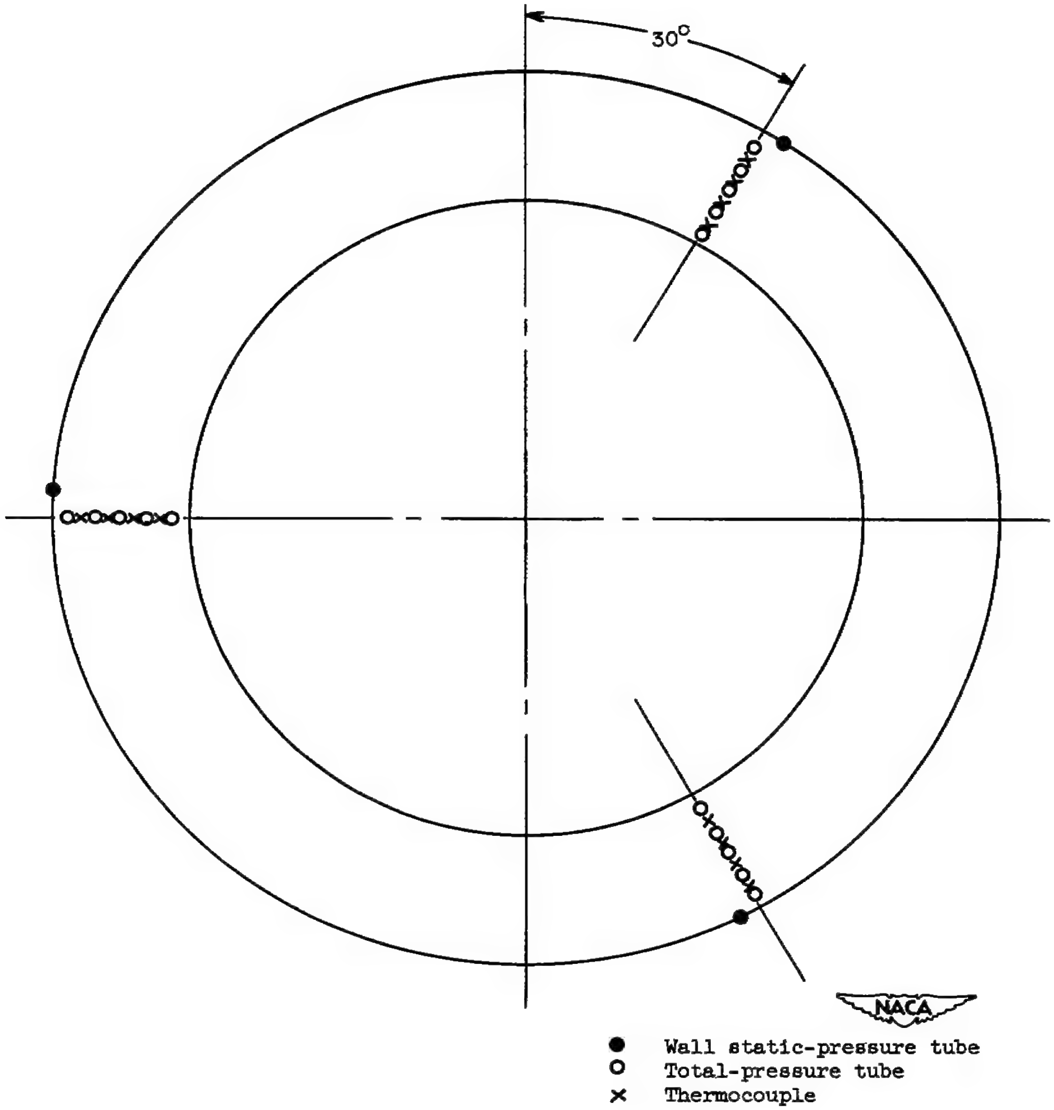
2625



(c) Instrumentation at turbine inlet, station 4, $1\frac{3}{4}$ inches upstream of leading edge of turbine-inlet guide vanes. Viewed from upstream.

Figure 2. - Continued. Instrumentation sketches of various measuring stations.

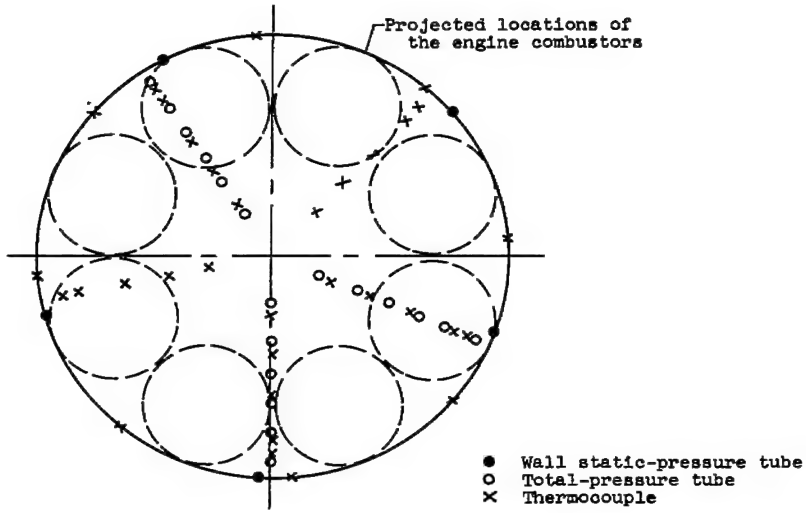
2625



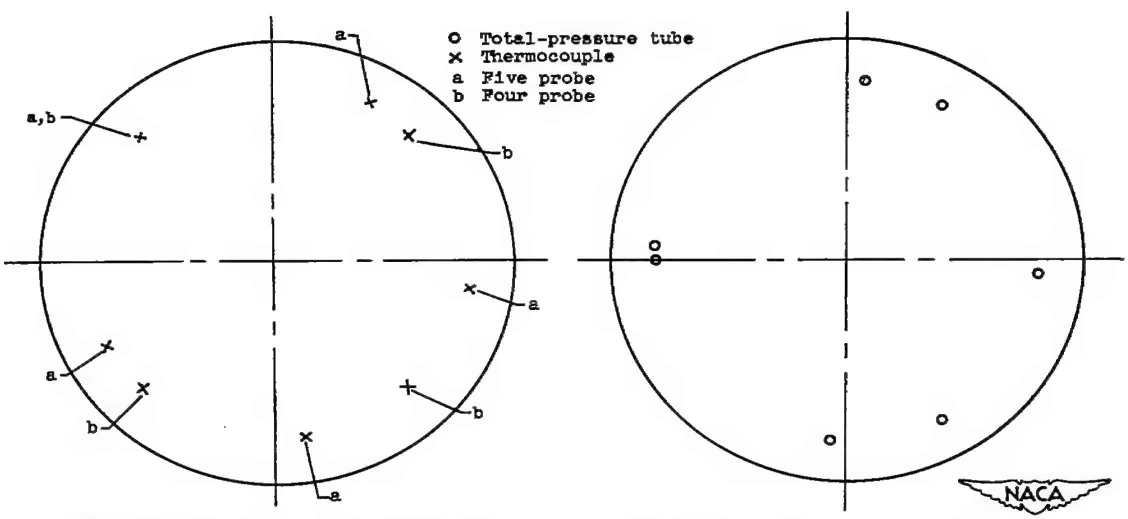
(d) Instrumentation at turbine outlet, station 5, $2\frac{3}{4}$ inches downstream of trailing edge of turbine blades. Viewed from upstream.

Figure 2. - Continued. Instrumentation sketches of various measuring stations.

2625



(e) NACA instrumentation at nozzle inlet, station 9, 15.15 inches downstream of tail-cone-outlet flange. Viewed from upstream.



Engine manufacturer's instrumentation

Air frame manufacturer's instrumentation

(f) Engine and air frame manufacturers' instrumentation at nozzle inlet, station 9, 15.15 inches downstream of tail-cone-outlet flange. Viewed from upstream.

Figure 2. - Concluded. Instrumentation sketches of various measuring stations.

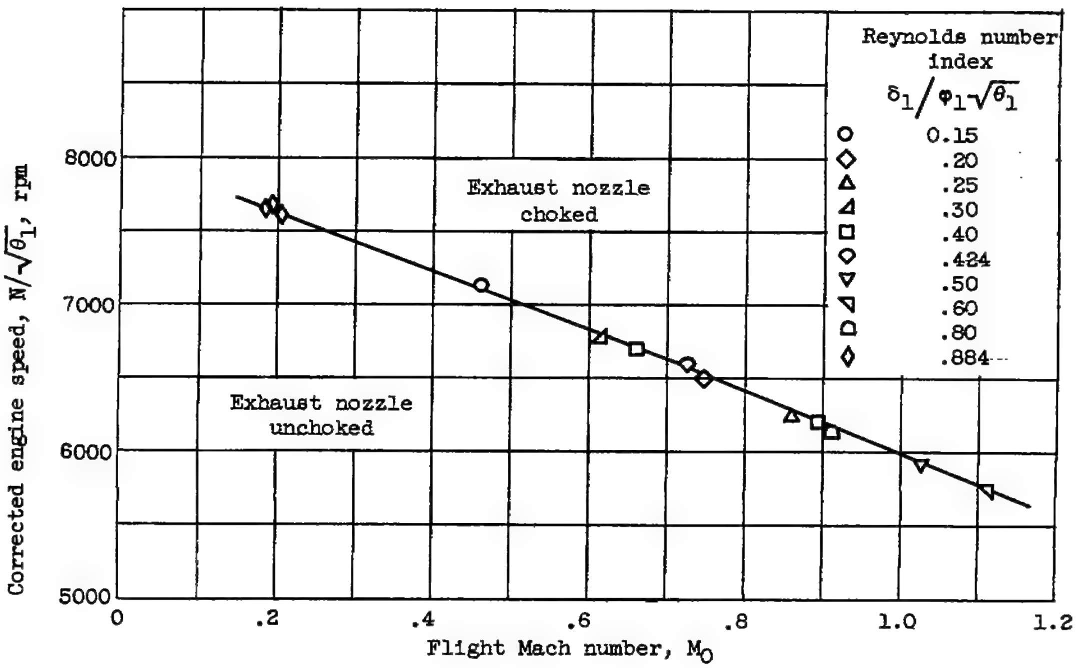


Figure 3. - Minimum corrected engine speeds at which critical flow existed in the exhaust nozzle.

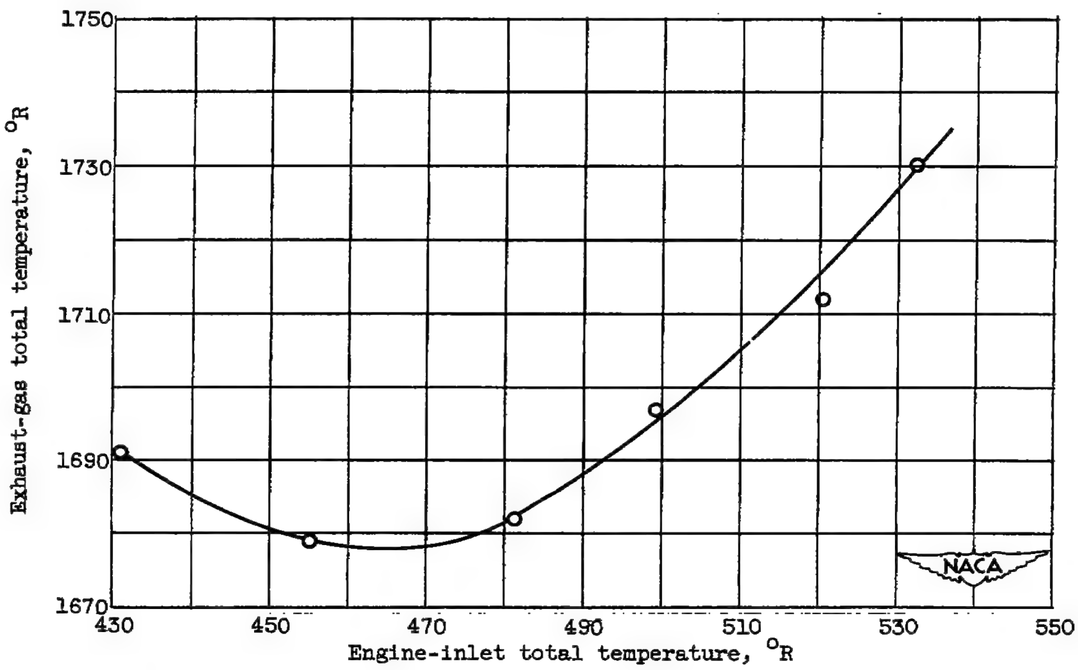
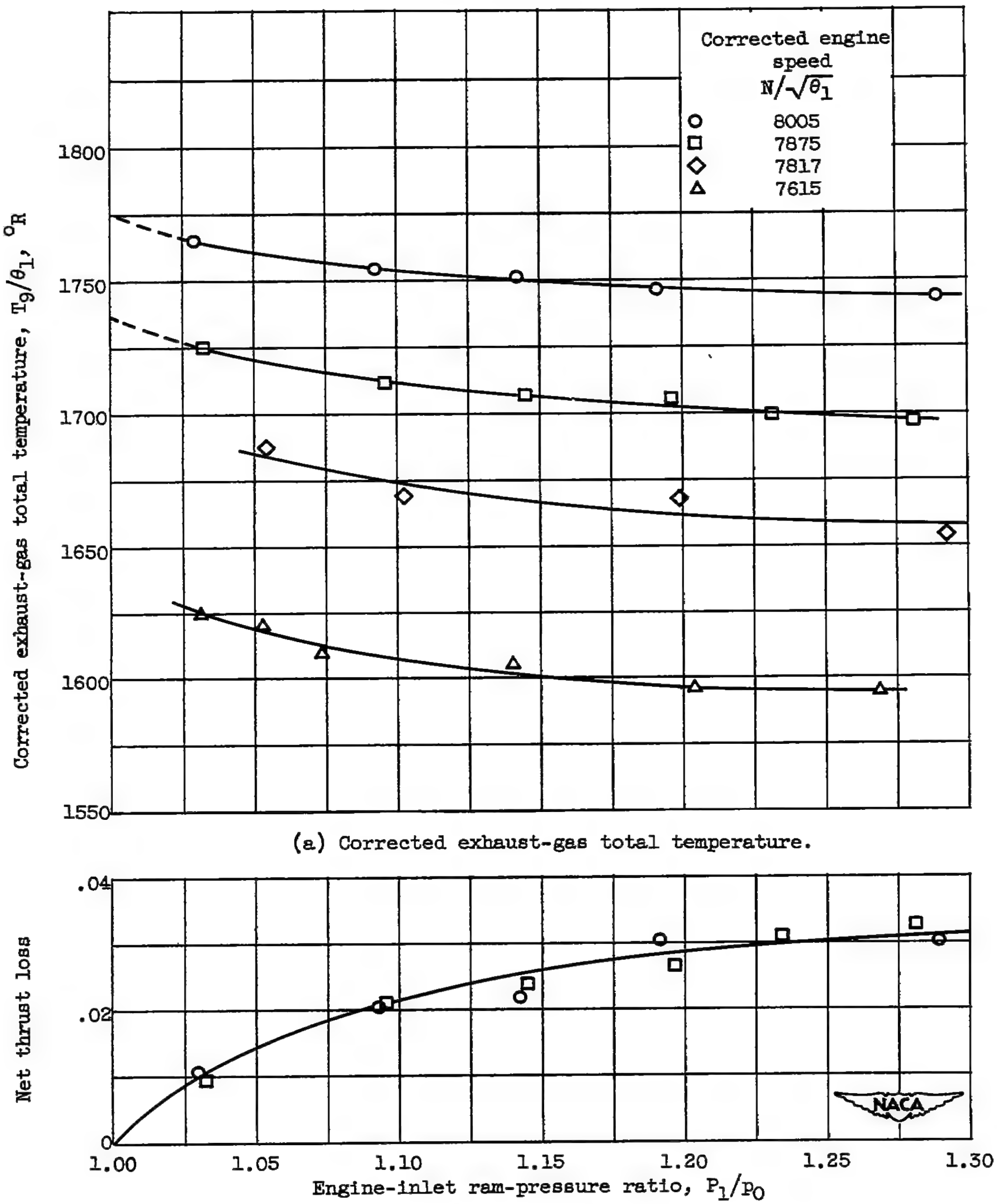


Figure 4. - Effect of engine-inlet total temperature on exhaust-gas total temperature. Engine speed, 7950 rpm; altitude, 20,000 feet; flight Mach number, 0.2.

2625

2625



(a) Corrected exhaust-gas total temperature.

(b) Net thrust loss.

Figure 5. - Effect of engine-inlet ram-pressure ratio on corrected exhaust-gas total temperature and net thrust loss for various corrected engine speeds.

2625

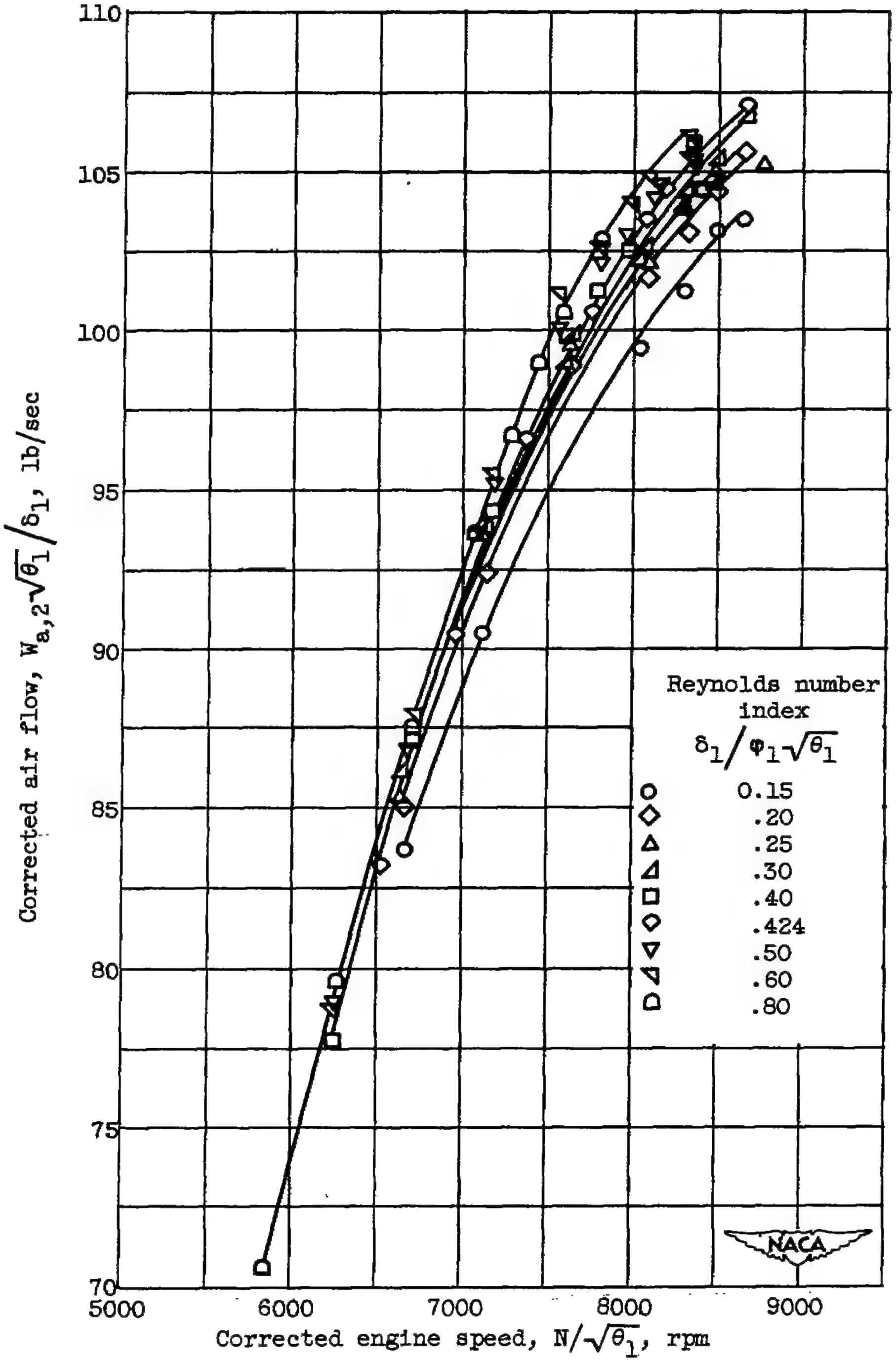


Figure 6. - Variation of corrected air flow with corrected engine speed for various Reynolds number indices.

[redacted]

2625

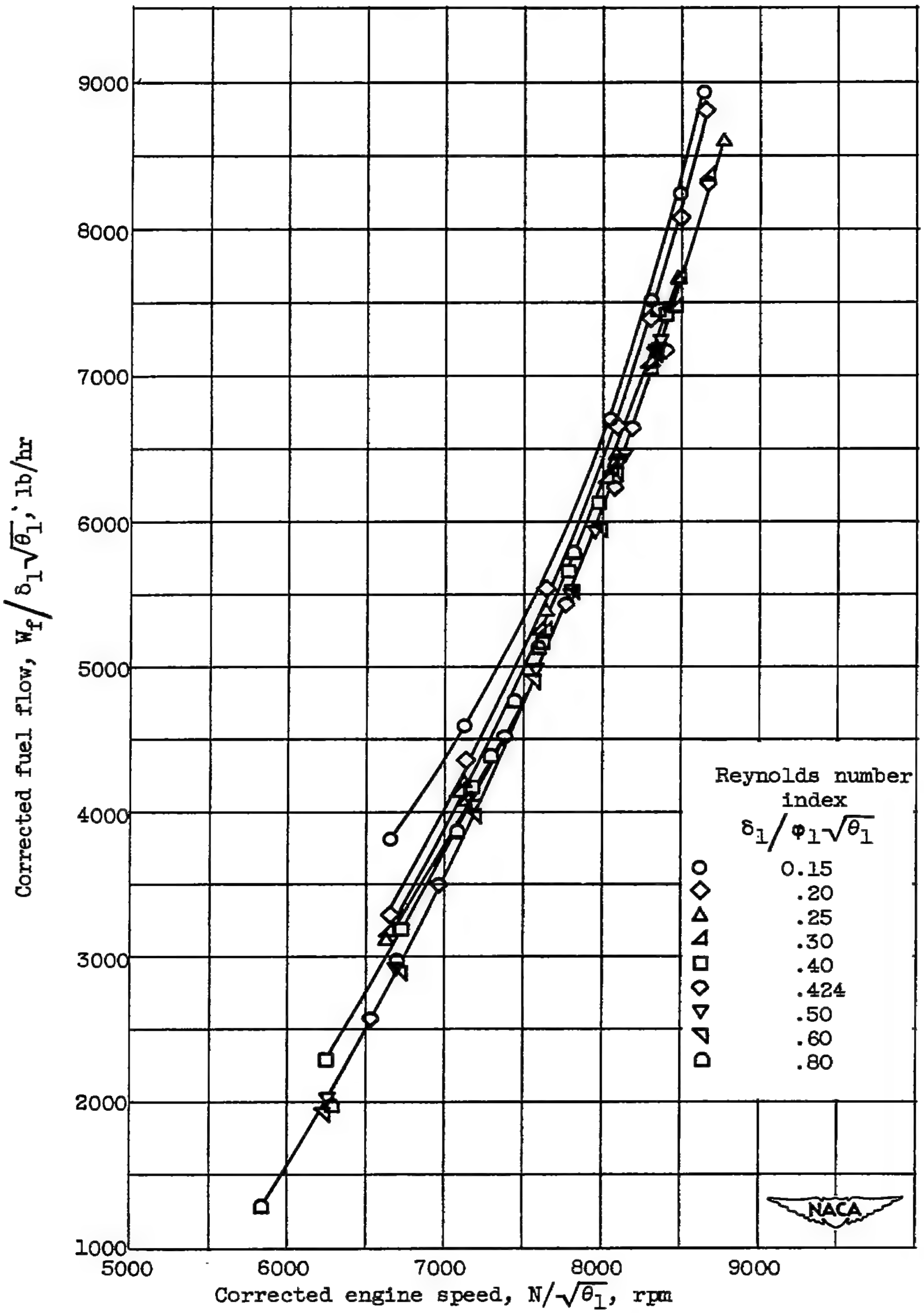


Figure 7. - Variation of corrected fuel flow with corrected engine speed for various Reynolds number indices.

2625

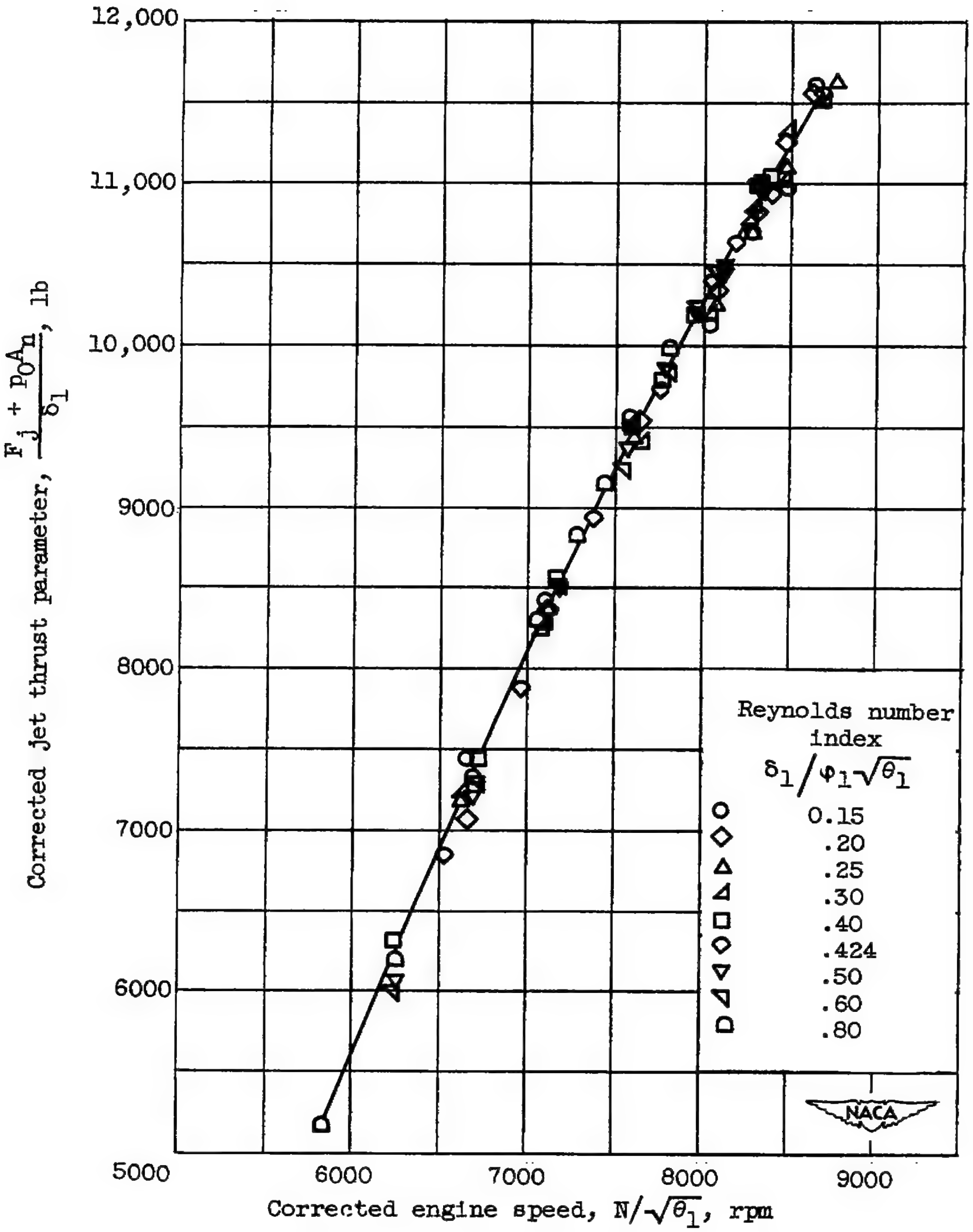


Figure 8. - Variation of corrected jet thrust parameter with corrected engine speed for various Reynolds number indices.

[redacted]

2625

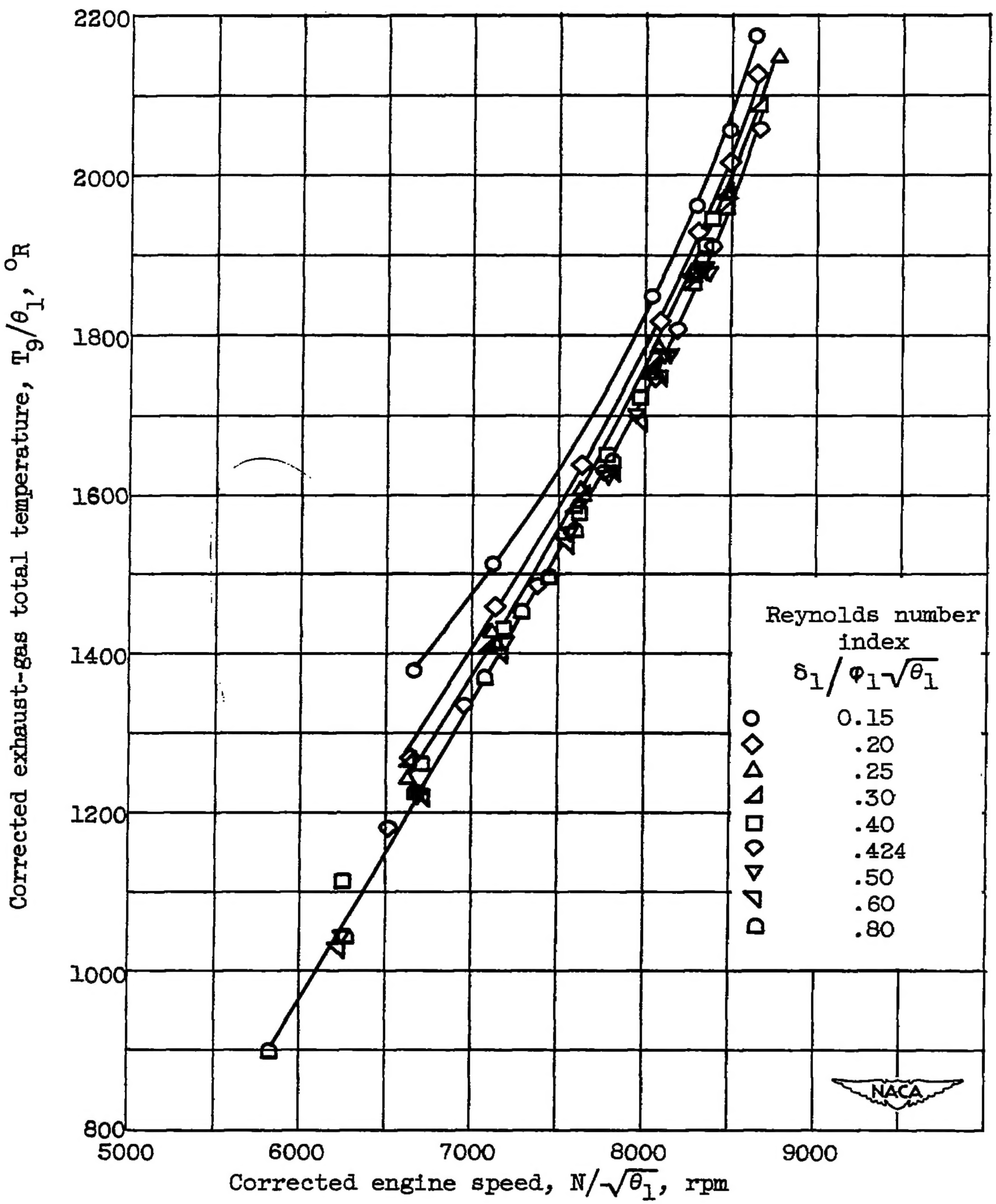


Figure 9. - Variation of corrected exhaust-gas total temperature with corrected engine speed for various Reynolds number indices.

2625

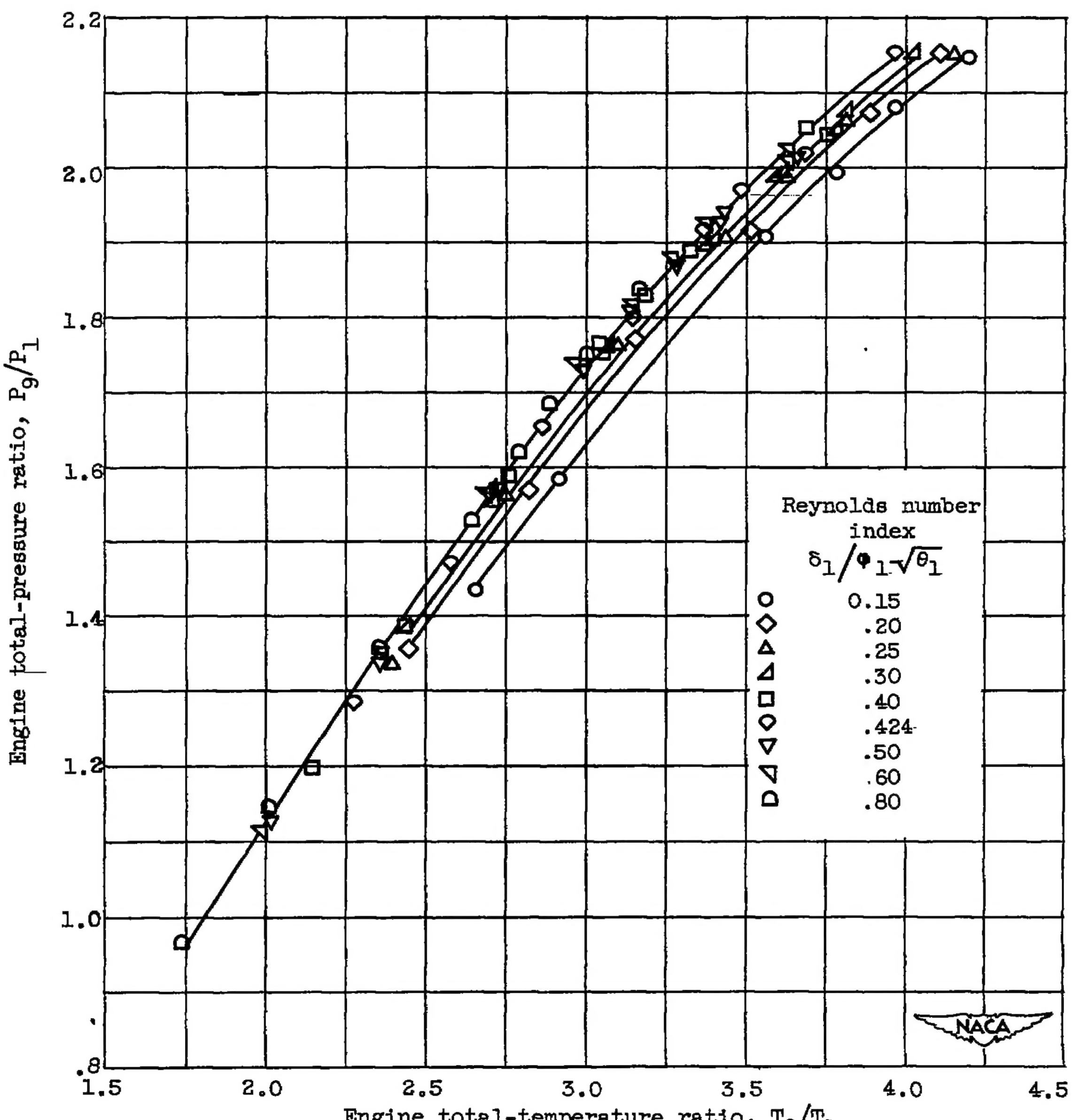


Figure 10. - Variation of engine total-pressure ratio with engine total-temperature ratio for various Reynolds number indices.

2625

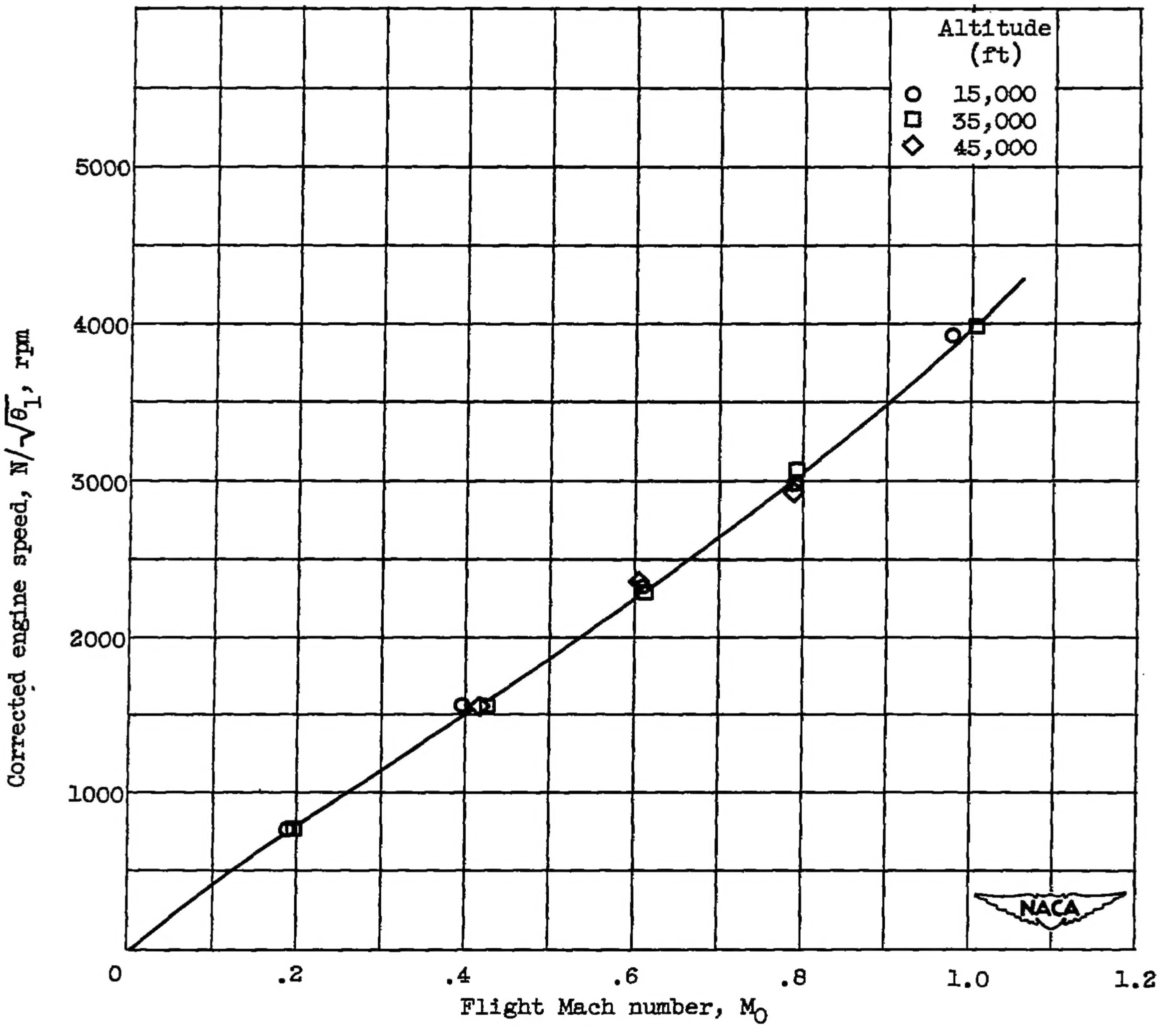


Figure 11. - Variation of corrected windmilling engine speed with flight Mach number at three altitudes.

SECURITY INFORMATION

[redacted]

[redacted]